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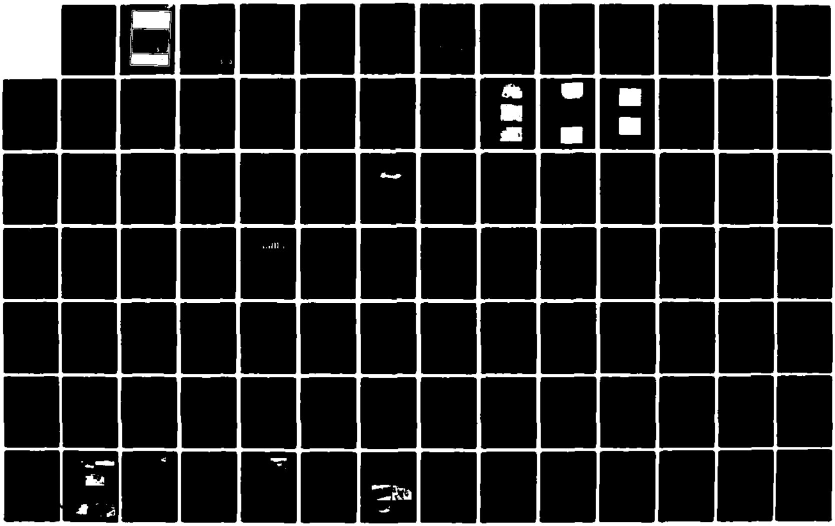

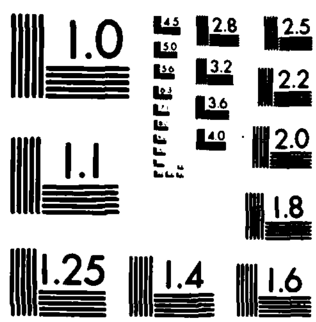
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ADVISORY GROUP FOR AEROSPACE RESEARCH & DEVELOPMENT

7 RUE ANCELLE 92200 NEUILLY SUR SEINE FRANCE

AGARD CONFERENCE PROCEEDINGS No. 347

Flight Mechanics and System Design Lessons from Operational Experience

NORTH ATLANTIC TREATY ORGANIZATION

DISTRIBUTION AND AVAILABILITY ON BACK COVER

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AGARD-CP-347

NORTH ATLANTIC TREATY ORGANIZATION

ADVISORY GROUP FOR AEROSPACE RESEARCH AND DEVELOPMENT

(ORGANISATION DU TRAITE DE L'ATLANTIQUE NORD)

AGARD Conference Proceedings No.347

FLIGHT MECHANICS AND SYSTEM DESIGN LESSONS FROM OPERATIONAL EXPERIENCE

Papers presented at the Flight Mechanics Panel Symposium held in Athens, Greece, 10–13 May 1983.

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Published October 1983

ISBN 92-835-0342-2

Printed by Specialised Printing Services Limited
40 Chigwell Lane, Loughton, Essex IG10 3TZ

PREFACE

One of the main fields of interest of the Flight Mechanics Panel is:

"Operational problems including such subjects as operational experience, safety, take off and landing, all weather operation and low altitude flight".

The Flight Mechanics Panel plays a unique role among the Technical Panels of AGARD by addressing the overall design of aircraft. This explains a continuous activity of this panel devoted to the lessons from operational experience on flight mechanics and design of aircraft and systems. This subject is covered by symposia at regular intervals, the first of which took place in 1970 in Baden-Baden, Federal Republic of Germany. It was entitled: "Lessons with emphasis on Flight Mechanics, from operating experience, incidents and accidents" (CP 70).

The following subjects were covered:

- use of recordings for routine flight operations
- rough field operations
- safety margins for take off and landing, high speed and low level flying
- accident surveys, including V/STOL and general aviation.

The second symposium was arranged in 1976 in Sandefjord, Norway, with the following title: "Aircraft operational experience and its impact on safety and survivability".

Again a first session was devoted to methods for accident statistics and analysis. The others were strongly oriented toward design (a subject which was not covered by the 1970 symposium):

- design practices for aircraft safety
- design for aircraft vulnerability and survivability.

Then a small session was devoted to aircrew considerations.

It can be seen from the description of the above two meetings that the second one put strong emphasis on the design aspect with little consideration of the man-machine interface.

From 1976 the major change which has occurred in the design of aircraft is the general use of electrical controls in combat aircraft (fly-by-wire) together with sophisticated systems used to provide man-machine interfaces, enabling an increase of efficiency of the crew, together with an increase of safety. Thus the periodic review of the lessons from experience which it was decided would be the subject for the spring 1983 meeting of the Flight Mechanics Panel was arranged in order to put some emphasis on these two major subjects.

The first session, usually devoted to the progress made in recording techniques, was reoriented toward the reporting systems used in various countries which exert influence on operations and design; the second session to adverse environment including rotorcraft icing, operation from high altitude airfields, windshear, atmospheric electricity. The third session was devoted to survivability after failure. Examples were given of new electrical control systems. The last session, the largest one with seven papers, was devoted to the man-machine interface.

Much has been learned from the discussions held after each presentation; it is hardly possible to sum up in a few words what could be the conclusions and recommendations about such a broad area covered by this symposium.

Nevertheless the following remarks can be made:

(1) detection of incidents, even minor ones, is a means of preventing accidents; accident prevention relies on a closed loop information system which must be usable by any crew (civil or military) and be anonymous in order to collect the right information without prejudice to any crew member. This is well shown by Session I with a special mention of the papers on incident reporting systems by NASA, and Air France which illustrate this point quite well.

(2) progress has yet to be made on knowledge of adverse conditions, weather conditions (sandstorms, windshears, icing) and definition of specific regulations for flying in difficult operational conditions (rough airfields, high altitude, high-slope runways etc...).

(3) the most controversial subject is that of the man-machine interface; great advances have been made in that area with the new head-up and head-down display and new control systems. But crews are always discussing the best means to use, the training needed to make proper use of new systems, the proving of new information systems.

On this subject the reader could begin with the stimulating paper by Farley entitled "Modern Flight Instrument Displays as a Major Military Aviation Flight Safety Weakness" followed by the entire session IV.

In conclusion, it can be said that such symposia are very useful as they provide a unique opportunity for discussion between aircraft designers and users. Thus, it is appropriate that another meeting of this type be arranged in a few years, when the emphasis could be on the subject of man-machine interface and the increasing use of digital computers and associated controls and displays, as they could influence the design of aircraft from an operational viewpoint.

L'un des principaux domaines d'intérêt de la Commission de Mécanique du Vol est défini comme suit:

"Problèmes opérationnels, incluant l'expérience opérationnelle, la sécurité, le décollage et l'atterrissage, les opérations tous-temps, et le vol à basse altitude".

En outre, la Commission de Mécanique du Vol joue un rôle unique parmi les Commissions Techniques d'AGARD en traitant de la conception d'ensemble de l'aéronef.

Ceci explique que la Commission exerce une activité continue dans le domaine des leçons de l'expérience opérationnelle sur la mécanique du vol et la conception de l'aéronef et de ses systèmes.

Ce sujet est donc traité par des symposiums tenus à intervalles réguliers.

Le premier eut lieu en 1970 à Baden-Baden, Allemagne Fédérale, sous le titre: "Les leçons de Mécanique du Vol tirées de l'expérience opérationnelle des incidents, des accidents" (CP 70).

Les sujets suivants furent traités:

- utilisation des enregistrements pour les opérations aériennes quotidiennes
- emploi des terrains irréguliers
- marges de sécurité à l'envol et à l'atterrissage, en vol à grande vitesse et basse altitude
- revue des accidents, incluant les V/STOL et l'Aviation Générale.

Le second fut organisé en 1976 à Sandefjord, Norvège; son titre était: "L'expérience opérationnelle sur aéronefs et son impact sur la sécurité".

A nouveau une première session fut consacrée aux méthodes d'analyse des accidents et aux statistiques correspondantes. Les autres furent principalement orientées vers la conception (sujet non traité au symposium de 1970):

- méthodes pratiques pour assurer la sécurité dès la conception
- conception orientée vers la sécurité et la survie.

Une petite session fut consacrée au problème de l'équipage.

On peut voir d'après la description du sujet des ces deux réunions que la seconde insiste plus sur la conception que sur l'interface homme-machine.

A partir de 1976, une révolution transforma la conception de l'aéronef: l'emploi généralisé de commandes électriques sur les avions de combat (commandes par fil) et en même temps, l'emploi de systèmes sophistiqués d'interface homme-machine, permettant d'accroître l'efficacité de l'équipage en même temps que sa sécurité.

C'est ainsi qu'il fut décidé de choisir la revue des leçons de l'expérience comme sujet de la réunion de 1983 de la Commission de Mécanique du Vol, afin d'insister sur ces deux sujets d'importance primordiale.

La première session consacrée dans les précédents symposiums aux progrès des techniques d'enregistrement, est réorientée vers les systèmes de comptes rendus utilisés dans divers pays pour exercer une influence sur les opérations et la conception.

La seconde session est consacrée à l'environnement adverse, incluant le givrage hélicoptères, les opérations sur altiports, les gradients de vent, l'électricité atmosphérique.

La troisième session est consacrée à la survie après panne et illustrée par des exemples des nouveaux systèmes de commandes électriques.

CONTENTS

La dernière session, la plus longue, avec sept communications, est consacrée à l'interface homme-machine.

Les discussions après chaque présentation ont été riches d'enseignements; il est difficile de résumer en quelques mots ce que peuvent être les conclusions et recommandations concernant le vaste sujet de ce symposium.

Cependant, on peut faire les remarques suivantes:

(1) la détection d'incidents, même ceux considérés comme mineurs, est un moyen de prévenir les accidents; la prévention des accidents repose sur un système d'informations doté d'une "boucle de retour", pouvant être utilisé par n'importe quel équipage (civil ou militaire) et être anonyme afin de collecter l'information sans porter préjudice à aucun membre d'équipage. Ceci est bien démontré par la Session I et en particulier par les communications sur le NASA Reporting System et sur le système employé par Air France, qui illustrent bien ce point de vue.

(2) Des progrès doivent encore être faits pour connaître les conditions météorologiques adverses (tempêtes de sable, gradients de vent, givrage) et pour définir des règles opérationnelles spécifiques pour voler dans des conditions opérationnelles difficiles (pistes irrégulières, pistes en altitude, à forte pente, etc...).

(3) Le sujet le plus controversé est celui de l'interface homme-machine; de grands progrès ont été faits à cet égard avec les nouveaux systèmes de présentation tête haute ou tête basse et les nouveaux systèmes de commande. Mais les équipages discutent toujours du meilleur système, de l'entraînement souhaitable pour l'employer, de la méthode pour en démontrer la validité.

Sur ce sujet, le lecteur du compte rendu du symposium pourra commencer par la communication de Farley intitulée "La présentation moderne des Instruments de Vol, faiblesse majeure de la Sécurité Aérienne des opérations militaires".

Après lecture de ce texte, le lecteur ne pourra faire autrement que de lire entièrement la Session IV.

En conclusion finale, on peut dire que de tels symposiums sont très utiles en donnant au constructeur une occasion unique de discuter avec les utilisateurs.

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AD P002695

INVESTIGATION, REPORTING AND ANALYSIS OF US ARMY AIRCRAFT ACCIDENTS

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SUMMARY

Each year aircraft accidents result in large losses of US Army equipment and personnel resources. This paper describes these losses in terms of aircraft and cause factors involved. Also presented is a system-oriented approach used in the investigation, reporting and analysis of these accidents. The results include identification of lessons learned with respect to cost, type aircraft, flight tasks, cause factors and system inadequacies which produced the cause factors.

INTRODUCTION

Problem. From fiscal year (FY) 78-82 the Army has experienced an average of 91 major[1] aircraft accidents per year. These accidents have produced an average of 27 fatal and 65 non-fatal injuries and $36.8M in costs per year. Figure 1 shows the cause factors of these accidents and it can be seen that human error is by far the largest. The problem for the safety community is to identify sources of these cause factors in the Army system, take corrective action and thereby make the system more efficient and effective.

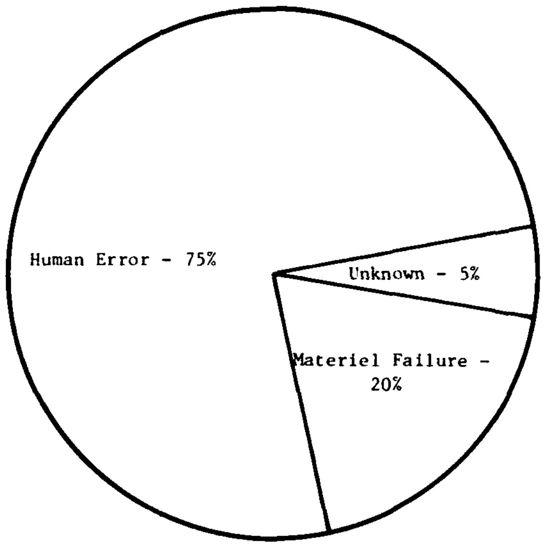

Figure 1 - Percentage of human error and materiel failure causes of major Army aircraft accidents from FY 78-82

[1]Class A,B, and C accidents costing more than $10,000 in injuries and property damage.

Approach. In the accident model presented in Figure 2 there are 12 items outlined which are basic elements of the Army system. When one or more of these elements is out of tolerance, an overload (item 13) is placed on the system role of personnel, materiel or the environment (item 14). When personnel or materiel in the system cannot cope with the overload, then personnel make errors, materiel fails/ malfunctions or the environment is allowed to have a negative influence on the performance of personnel or equipment (item 15). Most human errors, materiel failures and environmental factors do not lead to accidents (item 16). However, when circumstances operate unfavorably accidents occur (item 17).

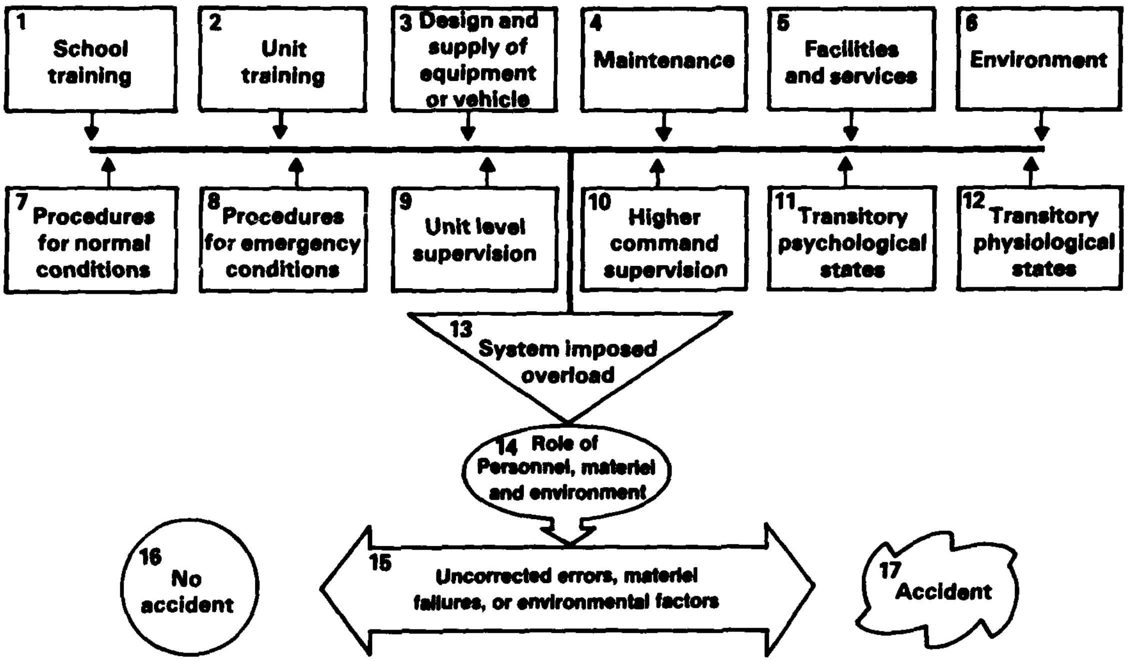

Figure 2 - Accident model

This accident model was used to develop an approach to accident investigation, analysis and prevention. Table 1 shows this 3W approach. It requires each investigator to answer what happened, what caused it to happen, and what to do about it with respect to man, machine and environmental cause factors. It can be seen that accident cause factors originate from inadequacies in the system. When remedies are applied which correct these system inadequacies, then human errors, materiel failures and the environment's affect on man and machine are reduced. As a result, fewer accidents occur and a more efficient and effective system is produced.

Table 1 - 3W approach to accident investigation analysis and prevention

WHAT HAPPENED	WHAT CAUSED IT	WHAT TO DO ABOUT IT
Human Errors Materiel Failures Environmental Factors	System Inadequacies	Remedial Measures

An example of the difference in investigating and reporting using the 3W approach can be seen by comparing the accident reports in Tables 2 and 3. Both reports are based on an OH-58 accident. Table 2 represents the type of informat on reported before implementation of the 3W approach. As can be seen, the investigation sto ed with t' .dentification of pilot error. The key question, "Why didn't the pilot use the landing l '+?" wa. ,t answered.

TABLE 2 - Example of information received on an OH-58 accident before the 3W approach was implemented

WHAT HAPPENED (Cause Factor)	WHAT TO DO (Remedy)
Pilot erred in that he did not use the landing light and allowed the tail rotor to strike tree branches during night landing.	Unit safety meetings should include a discussion of facts surrounding this mishap. Particular attention should be focused on the proper use of the landing light.
	Unit should also examine the adequacy of their night flying policy/training.

Table 3 shows the type of information reported using the 3W approach. It can be seen that the critical question of why the pilot did not use the landing light is answered through the identification of system inadequacies. Consequently, remedial measures in the third column address system inadequacies which are the root causes of the pilot error. When these system inadequacies are corrected, the overload is relieved, pilot errors are reduced and accidents from these errors become less likely.

Table 3 - Example of information received on the same OH-58 accident using the 3W approach

WHAT HAPPENED (Cause Factor)	WHAT CAUSED IT (System Inadequacy)	WHAT TO DO (Remedy)
OH-58A pilot...failed to perform a course of action required by TC 1-28, par. 5-28. He failed to use landing light to determine suitability of area for precautionary landing...	...Because required equipment is improperly designed. The landing light produced glare and obstructed vision of the pilot.	DARCOM redesign existing equipment to reduce unacceptable glare from OH-58A fixed landing light
	...Because of inadequate unit training. Pilot had not received sufficient training in execution of emergency procedures at night due to the unit being without an OH-58 IP for 3 months.	Unit Command upgrade unit training to maintain aviator night flight proficiency...

METHOD

The 3W approach was used to investigate 96 major accidents in FY 82. These accidents produced 45 fatal and 107 non-fatal injuries and a cost of $63M[2]. The Army Safety Center investigated all class A's and selected class B's and C's totaling 74 accidents. The remaining 22 accidents were investigated by field accident boards. The 3W information from reports of these accidents was then formatted into lessons learned and grouped by Aircrew Training Manual Tasks.

RESULTS

Table 4 shows the number and cost of lessons learned by type flight task and aircraft. Utility, observation and attack helicopters accounted for 82% of the lessons learned and 76% the cost. Human error accounted for 85% of the lessons learned and 74% of the cost while materiel failure accounted for only 15% of the lessons and 26% of the cost.

[2] This represents the worst annual record for Army aviation safety since FY 73.

TABLE 4 - Number and cost of lessons learned by aircrew training manual tasks and aircraft type

Aircrew Training Manual Tasks	Aircraft Types						Totals
	Observation OH-58 OH-6A	Utility UH-1 UH-60	Attack AH-1S	Training TH-55	Cargo CH-47	Fixed Wing OV-1D T-42	
Flight Planning	1* 1** $60***	2 4 $3,946					3 5 $4,006
Before-Flight Inspection		1 1 $648				1 1 $38	2 2 $686
Hovering	4 5 $380	1 4 $3,513	2 2 $105		1 1 $122		8 12 $4,120
Take off	2 2 $1,065	1 1 $1,444	1 1 $205	1 1 $34			5 5 $2,748
Basic Flight	1 1 $194					1 1 $3,190	2 2 $3,384
Approach and Landing	1 1 $248	6 9 $3,535	2 2 $2,480	1 1 $50	1 1 $43	3 3 $4,798	14 17 $11,154
Emergency	6 16 $2,223	2 5 $2,707	5 6 $5,154	1 1 $12		1 1 $36	15 29 $10,152
Tactical and Special	1 1 $483	3 6 $5,761					4 7 $6,244
Ground Taxiing					2 2 $122	1 1 $48	3 3 $170
Materiel Failures	2 2 $174	8 8 $6,794	1 1 $2,235	1 1 $26	2 2 $5,395		14 14 $14,598
Lessons Learned Number of Aircraft Involved Total Cost	18 29 $4,827	24 38 $28,348	11 12 $10,179	4 4 $123	6 6 $5,682	7 7 $8,109	70 96 $57,262

*Number of lessons learned
**Number of aircraft involved
***Cost multiplied by $1,000

Human error in two types of tasks accounted for almost half of all lessons learned; i.e., emergency tasks (21%) and approach and landing tasks (20%). These two tasks were also responsible for 37% of all costs.

Table 5 shows the distribution of the system inadequacies causing human error and materiel failure across flight tasks. It can be seen that inadequate supervision and inadequate self-discipline account for most of the system inadequacies. Examples of lessons learned are appended.

Table 5 - Number of lessons learned by aircrew training manual tasks and system inadequacies

Aircrew Training Manual Tasks	System Inadequacies									Total
	Written Procedure	Self Discipline	Supervision	Unit Training	Design	School Training	QC MFR	Maint	Other	
Flight Planning		4	1							5
Before Flight Inspection		2								2
Tactical and Special		4	3							7
Hovering	1	8	1	1					1	12
Take Off		3				1			1	5
Basic Flight		2								2
Approach and Landings		11	3						2	16
Emergency	4	7	9	1	7	1				29
Ground Taxiing		2		1						3
Materiel Failures			1		4		2	2	6	15
Total	5	43	18	3	11	2	2	2	10	96

DISCUSSION AND CONCLUSIONS

Aviation accidents from FY 82 were investigated and analyzed using the 3W approach. This enabled the identification of lessons learned with respect to cost, type aircraft, flight tasks, cause factors and system inadequacies that produced the human error and materiel failure cause factors.

The lessons learned identified human error as the primary cause factor, regardless of type aircraft, and these errors showed up mainly in emergency or landing and approach tasks. The most frequent system inadequacies leading to human error were inadequate self-discipline and inadequate supervision. The unit commander is in the best position to take actions that will correct these inadequacies within the unit. Therefore, all lessons learned from FY 82 have been compiled in a single report and distributed to commanders of aviation units. Additionally, the lessons learned report has been provided to major commands and higher staff levels for actions to correct system problems that are beyond the purview of unit commanders.

APPENDIX: EXAMPLES OF LESSONS LEARNED

TOPIC: TACTICAL AND SPECIAL TASKS

Lesson Learned #16: Aviators who intentionally violate written guidelines and verbal orders governing requirements of low level flight increase the probability of wire strikes.

Lesson Cost: Class A accidents: UH-1(3); OH-58
Fatal Injuries: 4
Non-Fatal Injuries: 15
Cost: $3,817,739

Problem: Because of excessive self-motivation or improper motivation, aviators intentionally fly their aircraft at low level altitudes and airspeeds that violate oral and written guidelines. As a result, aviators encounter flight problems from which they cannot recover, e.g., wires.

Corrective Action(s): Unit Commander take positive command action to insure aviators understand and comply with oral and written guidelines governing low-level flight altitudes and airspeeds and that these guidelines are not violated to enhance mission accomplishment.

TOPIC: NORMAL APPROACHES

Lesson Learned #37: Inadequate self-discipline of aviators adversely affects their ability to make sound decisions regarding by-the-book flight and safe aircraft operations during approach.

Lesson Cost: Class A accident: AH-1S
Class C accident: TH-55A
Non-Fatal Injuries: 1
Cost: $1,600,990

Problem #1: PIC's inadequate self-discipline (improper attitude) encouraged pilot to fly unautorized maneuvers which exceeded the pilot's and aircraft's abilities.

Corrective Action(s): Unit Commander take positive command action to inform personnel of problems encountered, monitor flight activity to identify improper attitudes/behaviors regarding by-the-book safe performance, and take effective enforcement actions to control problems identified.

Problem #2: Pilot was in a hurry to return to the heliport because of deteriorating weather conditions. In his haste the pilot reduced the throttle below necessary RPM (in contravention of TM 55-1520-238-10, para 8-43) causing him to land tail low.

Corrective Action(s): Recommend Unit Commander inform personnel of problems and remedies associated with these accidents. Include a discussion of the adverse affect "haste" can have on job performance.

TOPIC: FLIGHT PLANNING TASKS

Lesson Learned #13: A Unit Commander who fails to establish a crew rest policy in accordance with Table 5-1, AR 95-1 increases the probability of having fatigued aviators making critical errors.

Lesson Cost: Class A accident: UH-1
Non-Fatal Injuries: 2
Cost: $927,634

Problem: The lack of unit guidance regarding crew rest requirements for aviators promotes abuses in the assignment of work tasks for aviators, e.g., aviator continuing flight duties after having only nine hours of interrupted sleep in forty-eight hours.

Corrective Action(s): Unit Commander take positive command action to insure crew rest polices are established. AR 95-1, Table 5-1, may be used as a guide in this effort.

TOPIC: EMERGENCY TASKS

Lesson Learned #2: Failure to insure TM 55-1520-228-10 (OH-58) provides adequate instructions for describing tail rotor malfunctions and the correct emergency procedures for coping with them increases the probability of an aviator improperly handling this type of emergency.

Lesson Cost: Class A accidents: OH-58(2)
Fatal Injuries: 1
Non-Fatal Injuries: 3
Cost: $631,595

Problem: Tail rotor malfunctions and loss of tail rotor effectiveness has become a concern in OH-58 aircraft. However, about two pages of TM 55-1510-228-10 are delegated to a description of tail rotor malfunctions and emergency procedures. The terminology used is vague and the procedures described conflict from one paragraph to the next such that the reader is confused as to the required corrective action(s).

Corrective Action(s): DARCOM revise procedures in TM 55-1520-228-10B, Chapter 9, concerning tail rotor malfunctions such that corrective actions for each set of circumstances are expressed explicitly and without conflict.

AD P002696

THE USE OF FLIGHT RECORDERS IN THE INVESTIGATION OF AIRCRAFT MISHAPS

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SUMMARY

Some problems that have been encountered in the recovery of information from flight recorders for the investigation of aircraft mishaps are described. Techniques for rapid dissemination of the information to investigators are illustrated. Future developments in the design of voice and data recorders are discussed.

1. INTRODUCTION

Flight data recorders and cockpit voice recorders are now well established in the scenario of aircraft incident and accident investigation. Their unparalleled ability to provide detailed information on the events leading up to a mishap can rapidly indicate the areas in which the investigators should focus their activities.

In Canada, our experience with flight recorders has been confined mainly to transport aircraft. Ten years ago, we set up a Flight Recorder Playback Centre at the Flight Research Laboratory of the National Aeronautical Establishment to recover information from all voice and data recorders fitted to Canadian civil and military aircraft. The Centre is funded jointly by the National Research Council, Transport Canada, and the Department of National Defence.

The Centre is responsible for playback and analysis of information from all accidents and incidents as requested by the investigating authorities. Routine serviceability checks are also made of all the military recorder systems. Incident investigations have also been undertaken upon request from Canadian manufacturers, air carriers and the Canadian Air Line Pilots Association.

The centre is required to process fourteen different basic recorders and a number of minor variants. The civil systems involve separate voice and data recorders armoured and insulated to protect them in the crash environment. The voice recorders are designed to retain the last 30 minutes of information and the data recorders the last 15 to 25 hours. The majority of the military systems utilise a light-weight combined voice and data recorder housed in a deployable radio beacon otherwise known as a crash position indicator. The recorder retains the last 30 minutes of both voice and data.

A description of some of the capabilities and activities of the Centre was given at the Symposium on "Aircraft Operational Experience and its Impact on Safety & Survivability", held by the Flight Mechanics Panel at Sandefjord, Norway in 1976 (Ref.1). In this paper, I will attempt to summarise some of our subsequent efforts and comment on the future development of recording systems.

2. DATA RECOVERY

As described in Reference 1, from the outset our playback facilities were designed with emphasis on the recovery of information from damaged tapes and poor quality recordings. Our subsequent experiences, particularly with the first generation of digital recording systems, proved this decision to be a wise one.

Failure monitoring in these systems has a very limited capability indicating only whether the recorder itself is operational as opposed to checking the contents of the serial digital signal being recorded. Major defects in the signal have often remained undetected until a mishap has occurred. Such defects have included incorrect synchronisation codes, frozen bits in the analog-to-digital conversion, incorrect multiplexing of the parameters, and erroneous values due to erratic D.C. power supplies. Fortunately, in many cases, detailed study of the serial digital signal has permitted recovery of some, if not all, of the important information, though the task has sometimes been formidable.

The accident environment can involve abnormal levels of vibration and shock loading that affect the quality of the recorded signal through the mechanical limitations of the tape transport mechanism.

To assist in recovering information under these difficult conditions, we have developed hardware to cope with variable data rates. We also have an editing routine that allows us to examine the recovered serial digital signal for loss in bit synchronisation. We can then shift the bit sequence to re-synchronise the data. By examining high speed oscillograph records of the signal, we are also able to retrieve short periods of data that are too badly distorted for the playback system to decode them. These also may be edited into the data stream.

In our limited experience with catastrophic accidents, we have had to contend with recorders that have been immersed in sea-water, tapes that have been broken upon impact and last, but not least, fire damage.

Sea-water presents no major difficulty provided that the recording tape can be rinsed in fresh water immediately on removal from the sea-water to prevent salt deposits. We have had one case involving a metal tape where this was not done and experienced considerable loss of data until we finally managed to remove the resulting deposits.

We have not encountered any armoured recorders in which the tape transport has suffered impact damage, apart from one metal tape that fractured in two places. On the other hand, none of the accidents involved a high speed ground impact. The metal tape fractures were probably attributable to large angular accelerations associated with high rotary inertia of the tape on the reels.

We have lost three armoured and insulated recorders due to fire damage in two separate accidents. These units are required to be installed near the tail of the aircraft to minimise impact damage and are most frequently placed in the rear of the pressure cabin for ease of access. Whilst fireproofing specifications ensure that the units can withstand the temperatures of ensuing fires, prolonged immersion in the hot debris can result in over-heating of the recording medium, though there have been surprisingly few cases in which this has happened.

One of these accidents occurred to a twin engined jet transport when control was lost during an overshoot necessitated by the presence of a snowclearing vehicle on the runway. Although the crash site was within the confines of the airfield, deep snow prevented the only available fire vehicle from reaching the site promptly. The damaged flight recorders were not identified and removed from the wreckage until 20 hours later. When the charred remains arrived at the Playback Centre, we initially declared them both to be totally destroyed.

Figure 1 illustrates the condition of the flight data recorder. Aside from the metal components, only the remnants of the tape itself are evident, most of them being bonded to the metal cover. Figure 2 shows a close-up of the most important segments still on the tape transport with the recording heads and capstan on the right.

In the laboratory, we frequently use Soundcraft Magnasee to check on tape alignment. This consists of fine magnetic particles in suspension in alcohol. When we carefully applied this product to the charred tape, we found that the magnetic pattern was still there. We estimated that the tape transport had reached a temperature of 450°C.

With a packing density of only 151 bits/cm (384 bits/inch), the bit pattern made visible by the Magnasee (Fig. 3) could be read with a conventional microscope, though the fragility of the tape and numerous deposits on it made this a difficult and challenging task. We found that data on four of the eight tracks could not be deciphered due to an unidentified recording fault but that the accident information was fortunately on the good track that was nearest to the centre of the tape. This allowed us to read the data in spite of severe curling of the tape edges. Over a period of three weeks, we finally recovered 5000 bits of critical information covering three important 6 second periods immediately prior to the accident.

From a study of the wreckage, it was evident that, if the recorders had been installed in the tail-cone aft of the pressure bulkhead, very little damage would have ensued and the speed of the data recovery would have been measured in seconds rather than weeks.

Analysis of the recovered data also revealed a major fault in a synchro converter that affected seven of the parameters being monitored, but a detailed study permitted reasonable corrections to be made to most of the measurements. The information that we finally obtained proved invaluable to the investigation and more than justified the efforts expended.

Subsequent research has indicated that, with care, the Magnasee technique might be used to recover data from damaged tapes with packing densities up to 400 bits/cm (1000 bits/in). Unfortunately, more recent designs utilise packing densities as high as 817 bits/cm (2076 bits/in). For these tapes, we decided to investigate the use of a scanning electron microscope. Preliminary experiments have indicated that the data can be made visible with this technique (Fig. 4). For conventional magnetic tapes, excessive heating of the organic material by the electron beam can be avoided by use of low beam voltages and gold coating of the specimen that also prevents charge build-up on the tape surface. The one metal tape currently in use does not, of course, suffer from this problem.

In the case of our military deployable systems, the light weight beacon that contains the recorder has been designed to fly in a curved path after release to land outside the area of severe fire damage. There still remains a slight possibility that the tape may be damaged mechanically or by fire. Again, in this case, the packing density of the data on the tape is extremely low, partly as a result of the relatively high tape speed necessitated by the audio channels, and the Magnasee technique should be adequate.

3. DATA PRESENTATION

The other area in which we have made considerable progress is in the rapid presentation of the information in a meaningful form. The techniques that we have developed include plotting time histories of selected parameters in engineering units against appropriate time scales, reconstruction of perspective views of the aircraft as it moves along the flight path both in stored and real-time formats, and simulation of the aircraft instrument display and pilot's control inputs utilising the recorded data. These techniques are demonstrated in a short movie to be shown during the presentation.

We endeavour to maintain accurate directories specifying the parameters being monitored, their location in the appropriate serial digital format, and the calibrations required to convert the recorded digital numbers into engineering units for all Canadian civil and military aircraft equipped with recorders.

In any particular investigation, the calibrations may need modification depending on the availability of suitable corrections. When we convert the data into engineering units, we therefore store the directory used for the conversion ahead of the data stream. Selected parameters may then be printed out using a high speed line printer or plotted as time histories over the required time interval using a Tektronix 4014 computer graphics terminal. The parameter titles and units are picked up from the directory preceding the data. A Tektronix Hard Copy Unit is used to rapidly produce page-sized hard copies of the time histories (Fig. 5).

As far as the aircraft motion is concerned, print outs or plotted time histories of the altitude, airspeed, pitch, roll and yaw angles are not a very effective means of conveying a mental picture of the gyrations of the aircraft. We therefore utilise the recorded data to determine true air speed and, with the addition of the best estimates of the wind variations, compute the flight path coordinates. These are then combined with the recorded attitudes to generate the perspective views of symbolic aircraft drawn along the flight path as illustrated in Figure 6. The motion may be viewed from any fixed position in space. The one approximation that we normally have to make is to assume zero sideslip as this is not directly monitored by any of the data recorders. If sideslip could be estimated, it could of course be incorporated into the display.

On aircraft equipped with inertial navigation systems, accurate flight path information is directly available. In Canada, until the recent introduction of the Boeing 767 into service with Air Canada, the required parameters have not been recorded. We have not yet utilised this source of information.

An alternative presentation that we have developed has the symbolic aircraft flying across the screen of the computer graphics display in real time. We draw the reference grid in a horizontal plane (usually the ground plane) as in Figure 6 and have found it advantageous to include a shadow of the centre of gravity on the plane vertically below the aircraft. Where appropriate, cockpit voice recordings may be replayed in synchronism with the display.

This presentation can be generated on the Tektronix 4014 that is basically a storage CRT by using a "write-thru" mode. However, the display needs to be viewed in a darkened room. We have therefore acquired a Hewlett Packard 1351A Graphics Generator with a vector refresh display that produces a brighter image. The real time motion can then be recorded on film or video-tape together with the synchronised audio for dispatch to the investigators.

In search for more meaningful ways of presenting the wide range of recorded data, we originated the CRT re-creation of the flight instruments and controls as a poor man's substitute for feeding the recorder data into a flight simulator. One of our early efforts was presented in Reference 1 back in 1976.

We originally used the Tektronix 4014 displays with instrument scales and titles being written in the normal storage mode and the pointers or digital displays using the "write-thru" mode. More recently, the Hewlett-Packard display has been used to produce the brighter images for film or video-tape. Sample displays are illustrated in Figures 7 and 8.

As with the real-time display of the aircraft motion, the cockpit voice recordings can be replayed in synchronism with the display to provide a more realistic reconstruction of the situation on the flight deck. The display can be generated using either the original serial digital signal from the flight recorder or the subsequent computer tape format of the same information.

In the case of our military deployable systems, the recorder may be plugged directly into the playback system to immediately generate the combined audio-visual display. In a recent fatal accident that occurred to a CC-130 aircraft during a LAPES exercise, we delivered the preliminary information to the investigators in the field in the form of a video-cassette of this display within hours of receiving the recorder at the Playback Centre. It is an understatement to say that this was greatly appreciated.

With the civil systems, we normally re-record the serial digital signal onto one track of an audio tape and then record the carefully synchronised audio information onto the remaining tracks as on the military tapes. In the event that the recovered data

requires editing, we have a computer interface that allows us to re-generate the serial digital format from the edited data for synchronisation with the audio recordings.

The software required to generate these displays has been prepared for eighteen different aircraft/recording system combinations. It is flexible enough for rapid changes to be made to the display if this is desirable.

The displays use the sampled digital data directly and therefore show step changes in value at the sampling rate of the parameter involved. Although this feature is unrealistic, it does reflect the limitations of the information. It may also partially account for the one criticism of the displays that we have received from pilots, notably the lack of vertical speed indication. Until the introduction of the Boeing 767, this has not been monitored as a separate data parameter on Canadian aircraft.

It is gratifying to know that this display technique has since been adopted by other agencies and manufacturers.

4. VOICE RECORDING SYSTEMS

Current civil cockpit voice recorders retain the last 30 minutes of four audio channels including pilot, co-pilot, and third crew member channels and one for a cockpit area microphone. When there are only two crew members, the remaining channel may be used for recording passenger announcements. The crews' channels include all signals from both their microphones and headphones.

On many aircraft with a relatively quiet cockpit, conversation between the pilots does not necessitate use of their intercommunication system. In the event of a mishap, this conversation is only recorded on the cockpit area microphone channel. The level of background noise is, in many cases, sufficiently high to make the detection of some conversation difficult or impossible in spite of the use of advanced filtering techniques.

In the United Kingdom, where boom microphones are standard equipment, this problem has been largely overcome by requiring these microphones to be live to the corresponding pilot's channel at all times. As long as the crew members wear their head sets, high quality recordings of their speech are ensured and their seating positions are clearly indicated.

Unfortunately, on many civil transports in North America, hand-held microphones are still standard equipment. The live microphone technique is obviously irrelevant when they are in use. However, the aircraft normally have provision for the use of boom microphones as a crew option. With our limited experience, we would strongly recommend that these inputs be designed to ensure that they are live to the recorder at all times.

The use of the hot boom microphones does not eliminate the requirement for the cockpit area microphone as we are often interested in the background noises themselves.

With only the last 30 minutes of recording being retained, there have been many incidents or minor accidents following which the recorder has continued to operate for long enough to erase all useful information. To alleviate this problem, proposed changes to the ICAO flight recorder requirements include extension of the retention time to 60 minutes for transport aircraft over 27,000 kg. Such an extension has been technically feasible for a long time, but has been opposed by many pilots' associations who are concerned over mis-use of the information so obtained.

In some countries, these concerns have been well justified. From experience in Canada, it has been evident that protection from mis-use can only be guaranteed if appropriate legislation is enacted by the national government. The current trend towards freedom of public access to government retained information makes this imperative. Continuous detailed recording of the audio environment on the flight deck during accidents is a unique form of monitoring of human behaviour under extreme stress and certainly justifies some special consideration. It should be noted that, in the present day environment, even military aircraft operations are not necessarily isolated from these problems.

Some pilots associations have objected to the use of cockpit voice recordings for the investigation of incidents as opposed to accidents, though the information can still be of great value. It is hoped that these objections will be withdrawn if suitable legislation is enacted to avoid its mis-use.

The advantages of accurate voice/data synchronisation particularly in generating combined audio-visual presentations of the recorded information have already been discussed. The current civil requirement to include keying of all radio transmissions on the data recorder is not considered adequate as there may be no transmissions close to the time of the incident or accident. We have had the benefit of combined voice/data recording in accident investigations on both military and civil aircraft transports and recommend that the serial digital data signal should be recorded on one channel of the voice recorder as well as on the longer duration data recorder.

5. DATA RECORDING SYSTEMS

The design of the data recording systems for incident and accident investigation

involves a multitude of compromises that cannot possibly be covered in one short review, but there are a few aspects on which I would like to comment.

Due to the need to monitor many facets of the aircraft operation, the number of parameters being recorded is increasing as the aircraft become more complex. With the generally accepted requirement to retain the last 25 hours of data, the sampling rates that result are substantially lower than those normally used in flight test work and are often inadequate particularly in high speed mishaps. It has been frustrating to encounter many recording systems in which the recording capacity has not been fully utilised, and where, for the cost of a few extra wiring connections, the sampling rates of the more critical parameters could have been increased. The fact that the requirements specify only the minimum acceptable sampling rate has obviously not been adequately emphasised.

By the same token, in early designs in which special transducers have been installed for the recorder system, the impracticability of frequent calibrations of parameters in routine operations has resulted in the acceptance of much lower accuracies than would normally be specified for flight test work. Fortunately, the resolution of many of the parameters has been substantially better than the accuracy and, in the cases where post-incident calibration has been possible, acceptable data has been derived.

In later designs, greater use has been made of existing information sources in the aircraft such as the digital data bus. The accuracy and reliability of these sources are normally much higher than those of specially installed transducers as they are essential for operation of the aircraft. The use of these sources is therefore preferred provided that connection to the recorder system does not significantly degrade the reliability of the source.

Investigators have justified retention of the last 25 hours of data as a means of ensuring that data from at least one flight preceding a mishap is available for comparison with that obtained during the mishap as a check on erroneous data. This was certainly the case in an aborted take-off accident for which we were able to demonstrate from previous take-offs and landings that the special airspeed transducer for the recording system was in error by about 20 knots. When the recorder system is coupled to the more reliable and more accurate aircraft digital data bus, one wonders whether this argument for such a long recording period is still justified. The current Canadian civil requirement of 110% of the maximum flight duration might be more appropriate and would permit, for example, a doubling of the sample rate of all parameters for the same recorder memory capacity.

As mentioned earlier, over the years we have encountered a number of problems with the recovery of information from data recorders. These are mostly associated with mechanical imperfections e.g. tape mis-alignment, wow and flutter, particularly under abnormal conditions involving high levels of vibration. These imperfections are being reduced with improved design.

An alternative approach now being offered is the replacement of these units with solid state memories. These have the potential to provide substantial increases in reliability and an associated reduction in maintenance costs that look extremely attractive. Crash survivable modules have been demonstrated.

The major limitation of the units proposed so far is the restriction in memory size to only a small proportion of that currently provided by tape recorders. This is necessitated by economic constraints that hopefully will be eased as the technology develops. To minimise the effect of this limitation, it is proposed that data compression be used.

In routine flights, there are long periods when most of the parameters being monitored are almost constant. Data compression can be achieved by recording samples only when there is a significant change in value. This can greatly reduce the amount of information to be recorded. The reduction is, of course, tempered by the necessity of labelling each measurement with its associated time.

Various algorithms have been proposed in order to decide when values should be recorded. My one concern is that they will all reduce the resolution of the measurements, sometimes to an undesirable extent, in order to achieve worthwhile compression.

Unfortunately, most recorder requirements do not specify any required resolution, and it has been assumed to be synonymous with the specified accuracy. Again, only the U.K. civil requirements have covered this aspect, specifying that the resolution shall be five times better than the long-term accuracy. This will seriously restrict the degree of data compression that they can accept.

The solid state memory does have the advantage that it is more amenable to the storage of pre-selected sequences and events as opposed to continuous data. The last few minutes of data may be stored in uncompressed form or earlier events such as take-off, landings, or in-flight incidents retained. This capability can also partially offset the restrictions resulting from the limited memory size.

As regards the crash survivability, if the same level of protection is provided as for current recorders, we can anticipate that a few units will be damaged or destroyed. One would like to know whether slight damage is likely to cause total loss of all information or whether partial data recovery might still be possible as in the case of our

fire-damaged tape.

At present, it would appear that the solid state memory is best suited to aircraft with relatively low endurance such as helicopters and smaller military aircraft, though as the technology progresses it is conceivable that they may eventually replace all tape recorder systems.

6.0 CONCLUSIONS

The major contribution that flight recorders can make to the promotion of aviation safety has been well established. We would therefore like to see appropriate designs installed in a much wider range of aircraft types.

Apart from the immeasurable life-saving aspect of this contribution, the high replacement costs of modern aircraft can easily justify the expenditure necessary to install and maintain the recorder systems. This is particularly true on aircraft that have a digital data bus containing much of the required information.

We would also like to see a greater input from flight mechanics specialists into the detailed design of these systems to ensure that the information that they provide is adequate for the investigation of a wide spectrum of possible incidents or accidents.

REFERENCES

1. Caiger, B. The Recovery and Analysis of Accident Data from Flight Recorders in Canadian Transport Aircraft. AGARD Conference Proceedings No. 212, January 1977.

ACKNOWLEDGEMENTS

The contributions of D.F. Daw, M.G. Renton and S.J. Zurawski in the development and operation of the Playback Centre are gratefully acknowledged.

Figure 1. Fire Damaged Data Recorder

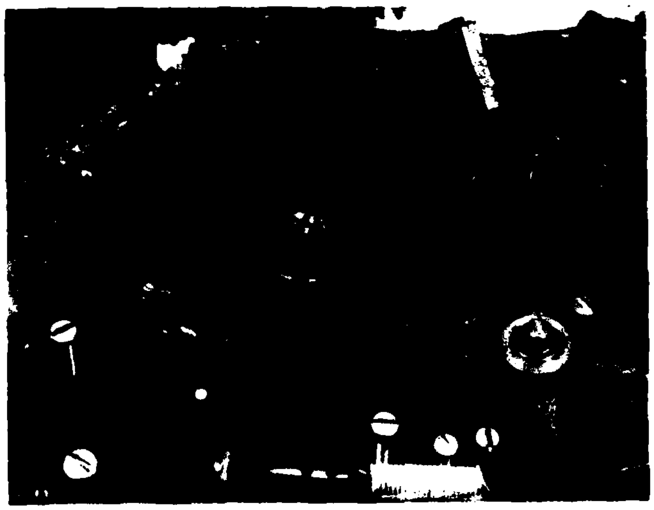

Figure 2. Charred Remnants of Data Tape

Figure 3. Digital Magnetic Patterns on Charred Tape

Figure 4. Digital Data from Scanning Electron Microscope

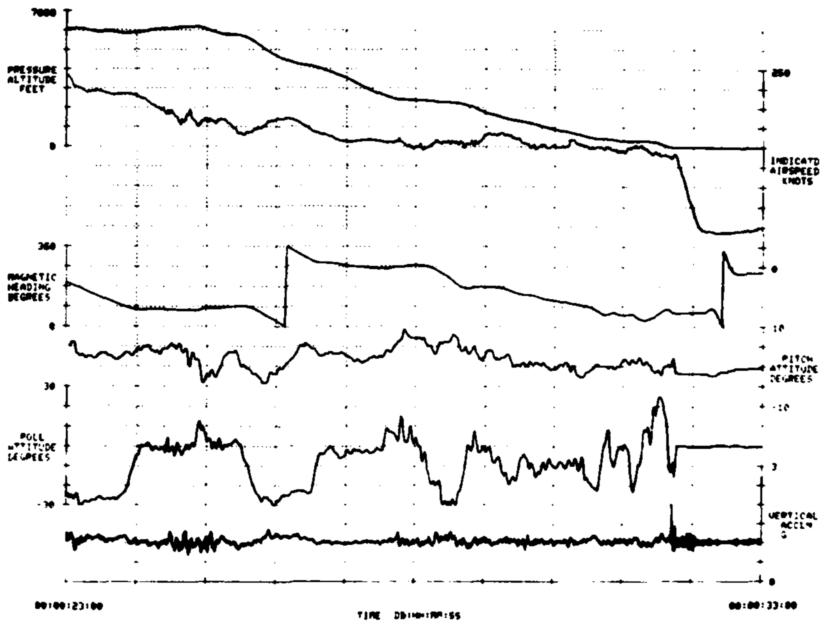

Figure 5. Typical Time History of Selected Parameters

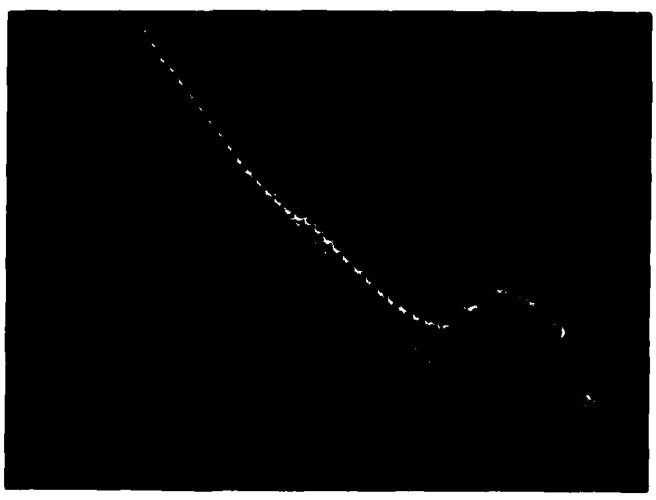

Figure 6. Presentation of Aircraft Motion

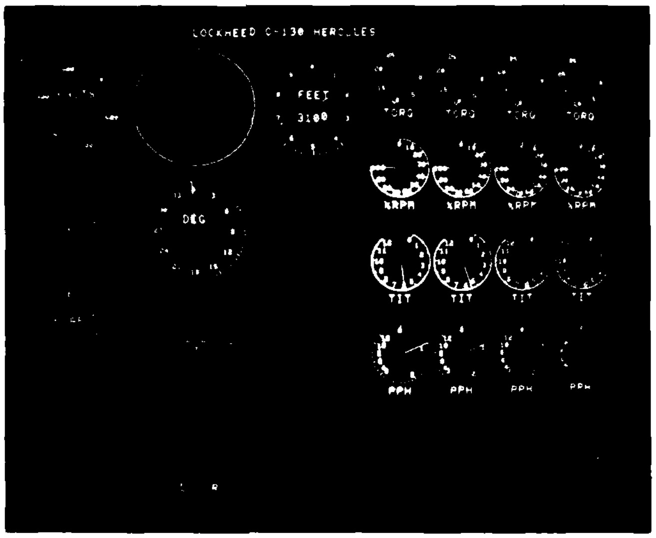

Figure 7. Lockheed C-130 Instrument and Control Display

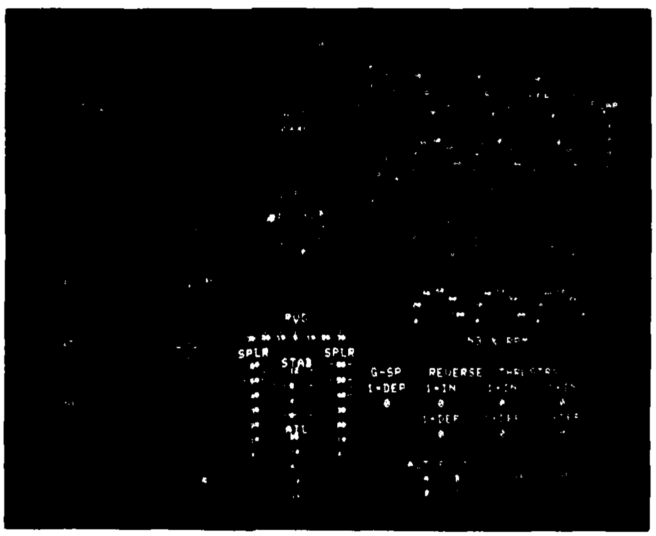

Figure 8. Lockheed L-1101 Instrument and Control Display

THE CIVIL AIRCRAFT AIRWORTHINESS DATA RECORDING PROGRAMME

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SUMMARY

The idea of operational flight data recording started with the continuous recording of just two parameters velocity V and normal acceleration g. The value of information gained from these simple recorders encouraged improved recorders and their development is traced up to the high capacity digital recorders available today. The data recording programme operated jointly by the CAA and the participating airlines has grown considerably over the 20 years of its existence. Its present organisation and activities are described, and a number of examples of the analyses performed on the data are given to illustrate the wide range of areas in which the data is useful.

1. Introduction

The safety of an aeroplane depends, amongst other things, on its being operated within a prescribed set of limitations. There are speeds above which it should not be flown and speeds below which it should not be flown, there are strength limitations which impose on the pilot a maximum manoeuvre 'g', and which define the maximum velocity of gust through which the aeroplane can fly without structural damage. To exceed any of these limitations represents a hazard. A hazard can also exist in other situations such as the achievement of excessive bank angle or descent rate close to the ground. We need to know at what level of probability these hazardous situations exist, and at what level of probability they may become extreme and cause, or contribute to an accident. It is a little surprising, therefore, that the practise of making measurements of the operational performance and behaviour of aeroplanes took so long to arrive, and that it is even now so limited in application. It is surprising because the design considerations which determine the strength and handling characteristics of an aeroplane are in many cases based on assumption, and if the aeroplane is to attain an acceptable level of safety and yet retain overall economy then these assumptions need to be verified. If they are too severe then the aeroplanes economy suffers, and if they are not severe enough then safety may suffer. It is true that many years of experience and satisfactory safety records have tended to justify some of these assumptions, at least to the extent that safety has not suffered.

This paper describes a programme of operational flight recording on civil aeroplanes which has been running in the UK for over 20 years, and is currently collecting data from about 170,000 hours of flying per annum on 7 different types of aeroplane. We call it the Civil Aircraft Airworthiness Data Recording Programme (CAADRP).

The principal objective, in a word, is safety. It is common experience that in nearly all accidents the investigation reveals that there were a number of contributory causes which individually would not have led to the accident. A poet who had never seen an aeroplane once said:

> Large streams from little fountains flow
> Tall oaks from little acorns grow.

It is the little acorns we are looking for to try to ensure that they do not grow into tall oaks.

2. Origins and History of CAADRP

The military aeroplanes which were in service at the beginning of the war in 1939 had been designed to relatively low manoeuvre envelopes, and it was the recognition of the fact that pilots under pressure in battle situations were quite likely to exceed the design manoeuvre envelope which led to the first application of operational recording with the V g recorder. This was a simple device which recorded the speed V and the normal acceleration 'g' continuously throughout the flight. These recorders, which could be left indefinitely in an aeroplane, scribed on a glass slide a continuous trace in the V g plane (Fig 1). If left in the aeroplane long enough, the total operational flight envelope would be drawn. By comparing the actual operational envelope with the design envelope it was possible to see the degree to which they were consistent. Data from a more recent report on some general aviation aeroplanes serves to illustrate the variations which can arise depending on the manner in which the aeroplane is flown, and the purpose for which it is flown. Figure 2 shows the V g record for an executive aeroplane flown by a professional pilot. The operation is contained well within the design envelope, the small area outside the envelope at the low speed end comes from the landing normal accelerations. Figure 3 shows a personal aeroplane, and here we see that the design diving speed V_D is often reached, and even slightly exceeded. Given that the maximum permitted speed V_{NE} is at least 10% below V_D, we see that the private operator

frequently exceeds the maximum permitted speed. He also produces higher landing loads than the professional pilot. In Figure 4 the envelope is handsomely exceeded with the aeroplane being used in the pilot instruction role, and one wonders why the aeroplane did not break, and in Figure 5 there is no boundary of the envelope which is not exceeded when the aeroplane is used for aerobatics. It can be seen that extremely valuable information was obtained from this simple recorder.

After the war the V g recorder was followed by the V g h recorder. This recorder also ran continuously and produced analogue traces of speed, normal acceleration and height on a paper band (Figure 6), and when the Boeing 707 came to England they were used to monitor the early operations of the first aircraft on the British register. An additional use of the data from V g h recorders on American civil aeroplanes was to provide a statistical picture of the atmosphere which formed the basis of the design gust requirements in both British and American airworthiness requirements, which still apply today.

The real beginning of CAADRP came in 1962 when the ARB (now the Airworthiness Division of the CAA) in conjunction with the Ministry of Aviation, and the Royal Aircraft Establishment set up a programme to install and operate special 12 channel recorders in a small number of aeroplanes of BOAC and BEA (now united as British Airways). These recorders used a light beam to produce a continuous analogue trace of each parameter on photographic paper. The output from these recorders were examined by CAA personnel looking for any occurrences which were out of the ordinary and would represent some erosion of safety margins. No specific rules were set out for this search, reliance being placed on the knowledge and imagination of the scrutineers. Occurrences detected in this way were called Special Events, and their safety implications were discussed with the operating airline.

The next major step came in 1966 when the carriage of crash recorders became mandatory in the UK. To comply with this requirement British Airways elected to fit digital recorders, but in addition, having been sufficiently encouraged by the results of the earlier programme, they decided to complement the crash recorder with a second recorder running in parallel but having a quick access cassette. The first aeroplanes to be equipped in this way were the Trident, but now nearly all British Airways aeroplanes have recorders with quick access cassettes. The change from 12 channel analogue recorders on only 5 aeroplanes to digital recorders having much larger capacity, on every aeroplane, magnified the data flow by an enormous factor, and made it quite impractical to search manually for Special Events. Fortunately one advantage of data in digital form is that it lends itself readily to handling by computer. The system which evolved has been operating since that time, and although there have been improvements both in the airborne recording equipment and the ground replay equipment, and in methods of handling the data, the essential features of the system have not changed and these are described later. Both British Airtours, and Gulf Air have similarly equippped aeroplanes and operate through British Airways. British Caledonian Airways have similarly equipped some of their aeroplanes and operate an independent programme, but all participating airlines cooperate on the discussion analysis and uses of the data.

3. Aims and Objectives

It is not sufficient to say, as I did earlier, that the objective is safety, it is necessary to indicate the means by which safety, i.e. the maintenance or reduction of the accident rate can be achieved. More specifically then the objectives are:

(i) Statistical validation of certification assumptions, suggesting remedial action where necessary (e.g. take-off and landing performances, adherence to operational speeds and limitations).

(ii) To detect unusual behaviour or situations which could be hazardous (e.g. large amplitude oscillation with Flight Management Systems engaged, effect of building induced turbulence).

(iii) To identify potential problem areas where safety margins may be eroded (e.g. probability of a tail or pod scrape as a result of cross wind landing techniques. Effect of changed procedures. Problems at particular airfields).

(iv) To provide back up data for specific investigations of incidents.

(v) To support research programmes (e.g. re-evaluation of gust statistics, low level wind shear, en-route vertical separation, vortex wakes).

(vi) To support work on development of airworthiness requirements.

4. Organisation and Administration

Since its inception CAADRP has been a cooperative venture, and although there have been many changes both in the way in which the system is operated and financed, and the roles performed by the various participants, we all still work together towards the same broad objective of safety. At present CAADRP is managed operated and financed by the airlines involved and the CAA, and this means agreement on the contents of the programme, its conduct and its cost. Another essential party to the agreement is BALPA (British Airline Pilots Association) whose interests are carefully safeguarded.

The largest contributor to the programme is British Airways with nearly all their aeroplanes:-

4	Trident 1
9	Trident 2
25	Trident 3
18	1-11 Type 510
3	1-11 Type 539
9	Tristar - 1
8	Tristar 200
6	Tristar 500 Active Controls
19	737
16	747-136
10	747-236
6	Concorde
2	757

British Airways also handle data from 7 Tristar 200's of Gulf Air and 9 Boeing 737's of British Airtours.

British Caledonian Airways have 4 DC10's equipped, and data from these aeroplanes is handled separately through a computer bureau.

Although the recording equipment used is not the same on all types of aeroplanes handled by British Airways, they all perform similar tasks and are dealt with in the same way. The aim is to record and replay every flight, but 100% coverage is not achieved for a variety of reasons. The duration of the cassettes varies with the type of operation. For the short haul operations the cassette duration is 14 hours, and for the long haul 50 hours. The cassettes are inserted into the recorder by ground engineers except in the case of the Tristar where the cassette is carried onto the aeroplane and inserted into the recorder by the crew. Additional cassettes are available to provide cover for all flying when the aeroplane is away from base for a long time. All cassettes are returned to the British Airways Flight Data Recording Section at Heathrow for replay. In the case of British Caledonian Airways the number of cassettes handled is limited, and the replay is performed by a commercial bureau.

Altogether, about 170,000 hours of flying is recorded annually and processed in CAADRP.

5. Technical Description

The 12 channel analogue records with which CAADRP started in 1962 were examined manually for undefined Special Events. With the change to digital recorders the number of parmeters recorded was greatly increased, and in addition to flight parameters which vary continuously such as speed, height, bank angle, heading, pitch altitude etc, it was possible to include discrete signals, such as gear up, gear in transit, gear down, which auto pilot or auto throttle was engaged and in which mode. A typical list is given in Appendix 1. The change to digital recorders also made essential the examination of the data by computer,and it was therefore necessary to define very precisely those Special Events which the computer was required to detect. Here the experience gained with the earlier analogue records was very useful. A working group with members from the airlines and CAA was set up to define an appropriate set of Special Events and these were written into a computer program SESMA (Special Event Search and Master Analysis). A typical list of Special Events is given in Appendix 2. The use of digital data introduced some difficulties, particularly where precise timing was necessary andthe loss of resolution due to a low sampling rate led to greater complexity in the program. The working group then met, and continues to meet, as required to resolve problems with the search program, to improve event detection methods, to introduce new events and to up-date the program in the light of experience. The program currently in operation is the 12th Version of SESMA. Data from various transducers on the aeroplane fed continuously, usually in the form of varying voltage, into a data management unit. Data from this unit was sampled in a fixed sequence and then converted into digital form before being fed onto the recorder. Not all parameters were sampled at the same rate, those that change slowly were sampled less frequently than those that changed quickly, the extremes being engine parameters which were sampled once every four seconds, on one aeroplane, and normal acceleration which was sampled at 8 times per second on all aeroplanes. Each cassette is subjected to a search by SESMA and when the computer detects a Special Event it automatically printed out an analogue trace of the most important parameters and discrete signals, about 20 in all, for a period from two minutes before the event till two minutes after the event. This print out included the scales for all the parameters shown, and additional information relevant to the flight, such as take-off and landing weights, weight at the time of the event, location of event etc. A typical output is shown in Figure 7. It is evident that the print out does not contain all the information present on the recorder, but this does not mean that any of the data is lost. It is always possible to replay a cassette and produce listings in engineering units, of all the data which has been recorded, in cases where the additional information is required for the study of a particular event. For this reason cassettes are not recycled immediately but held in store for about one month, and then recycled. It is the Special Events and the frequency with which they occur which forms the main basis of the analysis work.

The Special Event analogue, when it is first printed, contains enough information to identify the flight, so that additional information such as entries in the aeroplane Technical Log, records of the prevailing meteorological conditions, runway condition etc. can be made available in such cases where a detailed study of the event is necessary. All Special Events found are entered into a data bank which is held on disc. This data bank can be interrogated in a variety of ways so that statistics relevant to a particular type of event, or a particular type of aeroplane or even a particular airfield can be extracted. The information about each Special Event which is entered into the data base is deliberately made insufficient to identify the flight on which it occurred and hence identify the pilot. Similarly, those items of information which would enable the flight to be identified are eliminated from the analogue print out after a fixed time lapse. These actions ensure that none of the stored data can be associated with a particular pilot, and it can never be used to comment on any pilots performance.

A Flight Operation Working Panel in British Airways meets regularly to discuss event rates and trends revealed by the statistics, and where necessary it will discuss individual events of potentially special interest. Although the pilots anonymity is always strictly preserved he may at this stage be contacted for information by the BALPA representative on the Working Panel. It is emphasised that the intention of these enquiries is to seek a better understanding of an event which occurred, and in no way reflects on or comments on or accuses the pilot. Additionally the data is used within the airline in a number of other areas such as engine health monitoring, collection of data in support of autoland clearance, monitoring of data sources for the mandatory 'crash' recorder, etc. Detailed analysis of Special Events which forms the main part of the CAA activity is illustrated by some examples in the next paragraph.

6. Illustrative Examples

6.1 Turbulence:

One particular Special Event, airborne normal acceleration, has attracted much of our analysis effort recently, and has provided data which has been useful in a number of ways. Flight through turbulence is detected by the behaviour of the normal acceleration measured at, or close to, the centre of gravity of the aeroplane. Turbulence severe enough to produce large structural loadings was detected as a Special Event at an arbitrarily chosen excursion of ± 0.7 g. The detection method could not distinguish between g due to gust and g due to manoeuvre. However, since the printed analogue output for each event included other parameters, in particular, heading, bank angle, pitch attitude and some control surface positions, it was possible to say, from manual examination of the data, whether any manoeuvring took place and estimate its contribution to the normal acceleration excursion.

6.1.2 Combined Turbulence and Manoeuvre:

Although encounters detected at this level were relatively small in number some unexpected features were revealed. It was found that the percentage of occasions when turbulence and manoeuvre combined to accentuate the 'g' excursion was much higher than had been assumed. This fact was of particular importance for studies on active control aeroplanes where load alleviation and manoeuvre use the same control surface. Figure 8 illustrates a coincidence of the two 'g' sources.

6.1.3 Vortex Wake:

The second type of occurrence, Figure 9 was a very sharp excursion or series of excursion, of very short duration, whilst cruising in completely calm conditions at high altitude. The most probable explanation of such events was the encounter of the vortex wake of a leading aeroplane. Attention was drawn particularly to the event shown in Figure 9 because of its magnitude. An analysis of the event carried out by W. Pinsker at R.A.E. Bedford was able to deduce the intercept path with the vortex wake and explained the pattern of the 'g' and bank angle histories. Estimates of the asymmetric spanwise wing loading then indicated that very high wing root bending moments may have been reached. Another outcome of the R.A.E. analysis was the apparently very slow decay rate of vortices at high altitude. We have since been trying to collect data on similar incidents combined with radar records from which the flight paths of the two aeroplanes involved can be obtained. When a sufficient body of data has been collected it may be possible to make some estimate of the effect of altitude on the rate of decay of trailing vortices, and also on the probability of a severe encounter.

6.1.4 Distribution of Atmospheric Turbulence:

For the purpose of a special study undertaken by ONERA a much larger body of data was required, and to achieve this the detection level for the airborne normal acceleration Special Event was reduced to ± 0.5 g, which produced about 60 events per month. The ONERA work on the distribution of turbulence patches in the atmosphere is still continuing and will be reported elsewhere. One interesting feature revealed by the analysis so far was the very large difference between the gust probability deduced from the Boeing 747 data and that from the Lockheed Tristar. Since there was no reason to suppose that the atmosphere through which the 747 flew was any different from that flown by the Tristar, and since the recorded data was not suspect, an explanation had to be found in the analysis. In practice there is relatively little variation in the cruise speeds flown by the two aeroplanes, so for each aeroplane the g produced by a given gust will vary with aeroplane weight. If the weight at which a gust of given magnitude produces

0.5 g is high relative to the average weight then that gust will be seen more frequently than if the weight is low relative to the average. The difference in the 0.5 g cut off weights of the two aeroplanes explained the difference seen in the rate of encountering gusts.

The data also revealed that flight on some routes were more prone to turbulence than others, the extreme example being the Berlin corridor where aeroplanes were flying at 10,000 feet and inhibited from taking vertical or horizontal storm avoidance action. Here the frequency of gust encounter was very much greater than average, and individual aeroplanes which were used frequently on that route were subject to a much higher rate of fatigue damage.

6.1.5 Height Excursions:

Using the same set of recordings, data on height excursions in cruise has been extracted to provide a statistical picture of the probability of deviating from an assigned flight level. This data was prepared for a study on vertical separation standards. Figure 10 shows the probabilities, for one aeroplane type, of exceeding various height deviations derived from the turbulence data. It must be pointed out, however, that these probabilities must be an under estimate, as it is possible for quite large height excursions from the assigned height to occur without g exceeding ± 0.5. An example of such an excursion is shown in Figure 11.

6.1.6 Wind Shear on Approach:

This aspect of turbulence in which the headwind component, derived from the difference between calibrated airspeed CAS and inertia ground speed IGS, is examined for the presence of windshear is the subject of Mr. Woodfield's paper later in this symposium.

6.1.7 Monitoring of Lockheed Tristar with Active Controls:

The certification of the active control aeroplane required that certain assumptions regarding the behaviour and reliability of the active control system be verified in service by agreed analyses of recorded data supplied by British Airways to Lockheed. That verification work is not covered in this paper, but a direct measure of the effectiveness of the active control system was possible by comparing the behaviour of the aircraft in turbulence before and after active controls were fitted. Figure 11 shows the frequency of 'g' excursions for the two cases, from which it seems that a significant reduction in the aeroplanes response to turbulence has been achieved. However, the sample size is too small to allow firm conclusions to be drawn.

6.2 Abandoned Take-off:

The scheduled accelerate-stop distances which appear in the aeroplane Flight Manual, are based on measurements made during certification. In making these measurements a standard delay of one second is used between successive pilot actions following the decision to abort. This arbitrary delay time was chosen as being representative of pilot behaviour in an emergency stop, but has frequently been questioned and criticised for being too short. In order to get a measure of real operational delay times a study of high speed aborted take-offs was made. The study included only those take-offs which were aborted above 100 kts, since in low speed aborts with no great urgency to stop, the delays would not be representative. Fortunately aborts from high speed are relatively rare occurrences so that collection of data has gone fairly slowly. Data from 16 take-offs have been analysed so far, although in many cases the data is incomplete, so no clear picture is yet emerging, but it does seem that delay times as short as one second are not common. An interesting, although expected outcome from this work is the relationship between decision speed V_1 and the probability of aborting a take-off above 100 kts. Figure 12 shows the aborted take-off probability for a number of aeroplane types, and it can be seen that there is a large increase in probability as V_1 increases. This must be due, in part at least, to the longer time spent above 100 kts on the aeroplanes with the higher V_1s.

6.3 Scheduled Take-off Speeds:

Scheduled take-off speeds cater for the possibility of losing one engine and are governed by performance and handling considerations, and are further complicated by noise abatement procedures. Significant departure from these scheduled speeds erode the safety margins and represent a hazard, particularly in the case of speeds below the scheduled speed. In the case of the 747 it was found that there was a high frequency of 'speed high' events on the -136 and a high frequency of 'speed low' events on the -236. Apart from the engines the aeroplanes are identical except for the higher maximum weight to which the -236 is certificated. Investigation of these events revealed that because the stick forces at take-off were modified by the trim setting the heavier -236 had lighter stick forces for rotation with a consequent tendency to over rotate and hence become airborne with excessive pitch altitude resulting in a fall in speed. Once the source of the problem was recognised it was remedied by appropriate advice to the crews.

6.4 Speed Losses on Approach:

A significant loss of speed from the steady approach speed is detected as a Special Event, and it was noted that there was a large difference between the frequencies of occurrence of this event on the 747-136 and the 747-236. Since the same crews fly both aeroplanes the cause was looked for elsewhere. A probable explanation was found in the fact that the flight idle thrust of the RB 211 engines was much lower than that of the JT9, so that the 747-236 with RB 211s suffered a greater deceleration than the 747-136 when the throttles were closed. When the pilots attention was drawn to this difference, the problem disappeared.

6.5 High Altitude Buffet Margin:

At high altitude the measurement of maximum lift coefficient is determined by the maximum intensity of Mach buffet which the manufacturer is prepared to demonstrate. In normal operations the altitude is limited to ensure an adequate margin to the maximum demonstrated lift coefficient. Any diminution of this margin is detected as a Special Event. Our experience has been that this is a rare event, so when its frequency increased markedly on the L1011 there was a need to investigate the cause. The increase in frequency was found to coincide with the introduction of the Flight Management System, and on investigation it was found that this system had been incorrectly programmed and was flying the aeroplanes too high.

7. Limitations of Flight Recording

The flight recorder programme produced useful data in many areas, and a few examples have been quoted above, however, it must be admitted that quite exciting, and potentially dangerous events can occur without triggering any of the SESMA limits. To illustrate this point Figure 13 shows an upset to a Tristar in which an initial height gain was followed by a 4,000 feet height loss, and a descent rate of 6,000 feet per minute was reached.

This occurrence was brought to our attention by the Mandatory Occurrence Reporting System which operates in the UK and requires all personnel to report situations which have hazarded the aeroplane. It is quite rare to find any coincidence between Occurrence Reports and Special Events due to the difference between the two systems. The Special Event levels are generally set at a relatively low hazard level that would not justify raising an O.R. although sometimes the recorder does see situations which the crew may miss. On the other hand the Occurrence Reports, by definition, cover such a wide range of situations often involving a combination of factors that could not be defined in terms of a Special Event.

8. Future Work

8.1 Structural Loads:

In our present programme the only parameter giving an indication of structural load is the accelerometer at the centre of gravity. From work which has been conducted by the R.A.E. Farnborough on strain gauge measurements on a number of aeroplanes there are indications that large loads can arise in the structure during manoeuvres which are not accounted for in the design requirements. It would therefore be useful to record additional data to assess the magnitude of this problem in civil aeroplanes operations.

8.2 Flight Recording on Helicopters:

The increase in numbers and passenger carrying capacity of helicopters, and the fact that their accident rate appears to be higher than that of fixed wing aeroplanes all point to the need for a better understanding of helicopter operations. To this end the possibilities of recording operational data on some civil helicopters is being actively pursued.

APPENDIX 1

Aircraft Type: B737-2326

System Type: PVS1940

Recorder Type: QAR/DFDR

VARIABLES

ACRONYM	PARAMETER	UNITS
ALT	Pressure altitude (combined coarse and fine)	feet
ALTC	Pressure altitude - coarse	feet
ALTF	Pressure altitude - fine	feet
CALI	System calibration word	units
CPP	Control column position - pitch	degrees
CPR	Control wheel position - roll	degrees
DOC	Data management entry panel data (AIDS panel)	octal
EPR	Engine pressure ratio Engine Nrs. 1, 2	units
FF	Fuel flow rate Engine Nrs. 1,2	kgs/hr
FLAP	Flap - trailing edge position	degrees
GMT	Greenwich mean time	hrs/mins
GS	Glideslope deviation 75μA = 1 dot	μA += above G/S (fly down)
GW	Gross weight	kgs
HDGM	Heading - Magnetic	degrees
IAS	Indicated airspeed	knots
LATA	Lateral acceleration	g units
LNGA	Longitudinal acceleration	g units
LOC	Localiser deviation 75μA = 1 dot	μA += left (fly right)
N2	N2 spool RPM Engine Nrs. 1,2	% RPM
NMLA	Normal acceleration (not corrected for datum errors)	g units
PITCH	Pitch attitude	degrees +=nose up
RALIH	Radio height Nr. 1 (high range)	feet AGL
RODB	Rate of descent, barometric (AIDS computed value)	feet/min
RODR	Rate of descent, radio (AIDS computed value)	feet/min
ROLL	Roll attitude	degrees +=right wing low
RUDP	Rudder pedal position	degrees
SBLP	Speed brake lever position In armed detect = 4.0 deg) In flight detect = 40.5 deg) nominal values Full up = 51.0 deg)	degrees
STAB	Stabiliser position	units of trim 0 = 3 deg L.E. up 17 units = 14 deg L.E. up
SUBFRM	Subframe number and position of second in 4 second data frame	
TAT	Total air temperature	degrees C

DISCRETES

ACRONYM	PARAMETERS	STATUS
AACQ	Altitude Acquire	1=acquire
AFW	APU fire warning	1=warning
AHLD	Altitude hold	1=hold
APAM	A/P engage A CMD	1=selected
APBM	A/P engage B CMD	1=selected
APAW	A/P engage A CWS	1=selected
APBW	A/P engage B CWS	1=selected
APPR	Approach	1=selected
AOW	APU overheat warning	1=warning
ATE	Autothrottle engage	1=engage
ATME	Autothrottle mode _ EPR	1=selected
ATMP	Autothrottle mode - PDC speed	1=selected
ATMS	Autothrottle mode - IAS	1=selected
DFDR	DFDR status flag	1=fail
EF	Engine fire warning Engine Nrs. 1,2	1=warning
EO	Engine overheat warning Engine Nrs. 1,2	1=warning
FA	Flare armed	1=armed
FE	Flare engage	1=engaged
GA	Go around	1=pressed
GDW	Glideslope deviation warning	1=warning
GLD	Gear lever down	1=down selected
GPWS	Ground proximity warning	1=warning
GSE	Glideslope engage	1=engaged
FDAU	FDAU unit status flap	1=fail
HE.A	Hydraulic system A Engines Nrs. 1,2	1=warning
HP.B	Hydraulic system B Pump Nrs. 1,2	1=warning
HP.O	Hydraulic system pump overheat Engine Nrs. 1,2	1=overheat

ACRONYM	PARAMETER	STATUS
HQB	Hydraulic system B - quantity	1=low
HQS	Hydraulic system - stand-by quantity	1=low
LCHG	Level change	1=selected
LDW	Localiser deviation warning	1=warning
LF.E	Leading edge flap - extended Section Nrs. 1, 2,3 and 4	1=full extend
LF.T	Leading edge flap - in transit Section Nrs. 1, 2, 3 and 4	(1=intransit (0=locked in
PDCE	P.D.C. engage	1=engaged
PEV	Pilot event marker	1=pressed
PW	Pressurisation warning	1=auto-fail
RT.E	Reverse thrust extend Engine Nrs. 1, 2	1=extend
RT.U	Reverse thrust unlock Engine Nrs. 1, 2	1=unlock
SL.E	Slat - full extend Positions 1-6	1=full extend position
SL.M	Slat - mid extend Positions 1-6	1=mid extend position
SL.T	Slat - in transit Positions 1-6	1=in transit
SQSW	Squat switch	1=made
STD	Stabiliser trim - DOWN	1=activated
STO	Stabiliser trim - UP	1=activated
VHF	VHF keyed Nrs. 1,2	1=press to transmit
VLA	VOR/LOC arm	1=armed
VLE	VOR/LOC engage	1=engaged
WWF	Wheel well fire	1=warning

APPENDIX 2

EXAMPLE OF SPECIAL EVENT DEFINITION

EVENT		A/C	DETECT LEVEL	ALERT LEVEL	DATA BASE MESSAGE
8C	Climb out speed low, 35 ft AGL to 400 ft AAL	747	V2/1 sec	As detect	Low: V2-(A-V) at Height' AAL
		737, TS) 1-11,TRD)	V2 + 5kts/5 secs	V2/1 sec	
		CONCD	V2/3	V2/5 secs	
8D	Climb out speed low, 400ft AAL to 1500ft AAL	CNCD,	V2 + 15kts/3 secs	As detect	Low: V2-(A-15-V) knots at HEIGHT' AAL
		TS,TRD,737	VNA - 10kts/5secs	V2/1 sec)	
		747	VNA - 10kts/1 sec	V2/1 sec)	
	(CNCD to 15000ft)	1-11	VNA - 15kts/5 secs	V2/1 sec)	V2-(A-V)
		Gulf Air		)	
		TriStar	V2/5 secs	V2/1 sec)	
9A	Pitch rate high take-off (between 4 deg & end of rotation)	737 only	5.5 deg/sec	As detect	(Max) High: PRM degs/sec
9B	Pitch rate (average) low during take-off	737 only	1.5 deg/sec	As detect	Ave.Low: PR degs/sec
10A	Unstick speed high	TS	V2 + 18kts	As detect	V2 + V-(A-18) knots
		1-11,TRD 747,CNCD	V2 + 15kts	As detect	V2 + V-(A-15) "
		737	V2 + 15 + Bkts Buffer value B = 0.7(50000-TOW)1000 (but not -ve)	As detect	V2 + V-(A-15-B) knots
10B	Unstick speed low	CNCD	V2 - 10 kts	V2 -12 kts	V2 - (A+12-V) knots
		Other a/c	V2 - 5 kts	As detect	V2 - (A+5-V) "
		1-11	V2 - 5 kts	V2 - 8 kts	V2 - (A+8-V)
10C	Tyre limit speed high at T/O	CNCD only	VTYRE	As detect	Limit + V-(A-2) knots IGS

CAYABN

FIGURE 1

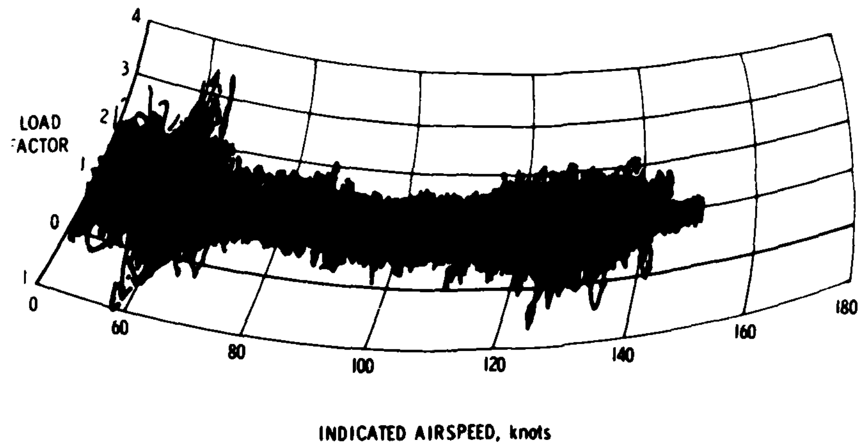


FIGURE 2

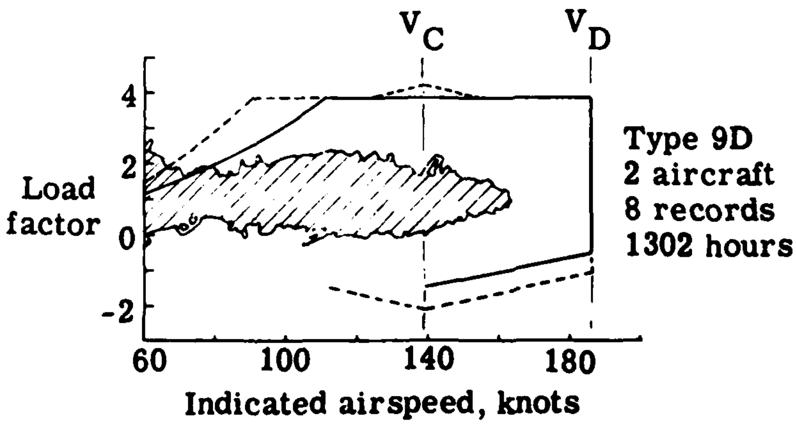


FIGURE 3

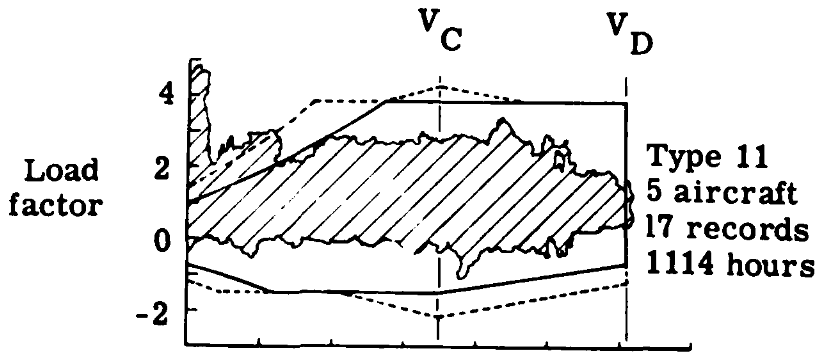


FIGURE 4

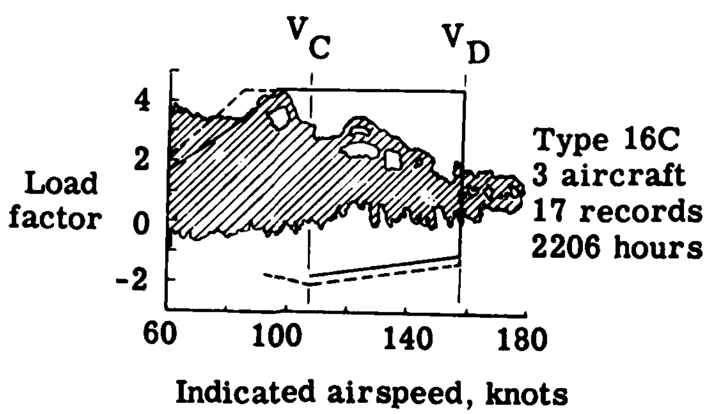


FIGURE 5

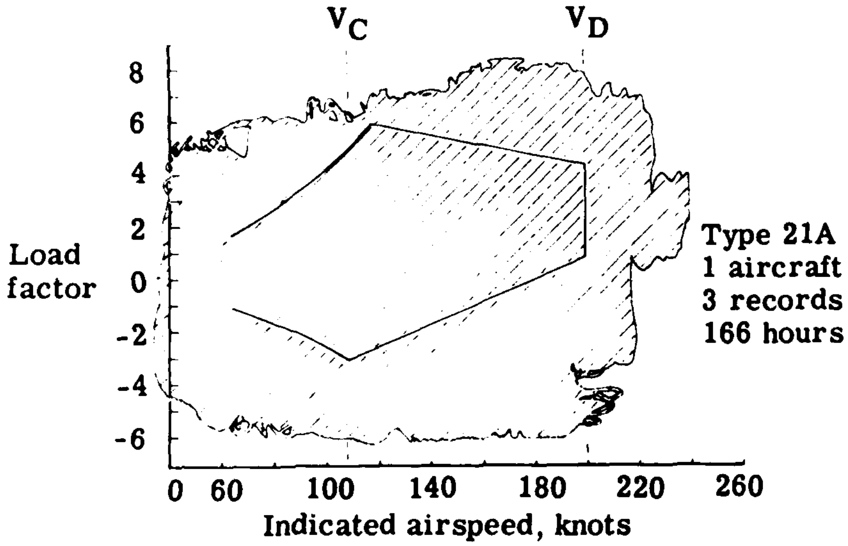


FIGURE 6

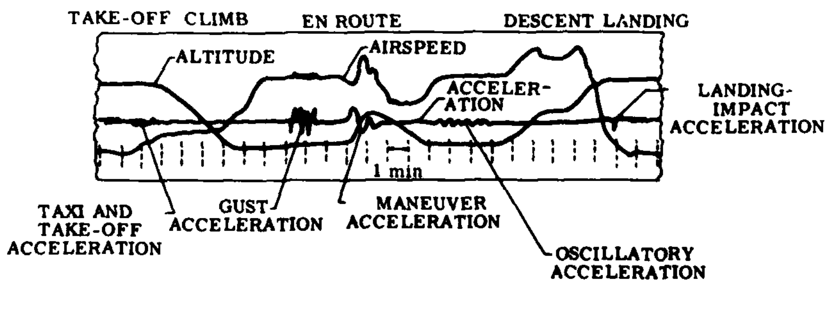


Figure 3.- Illustrative VGH record.

Figure 7

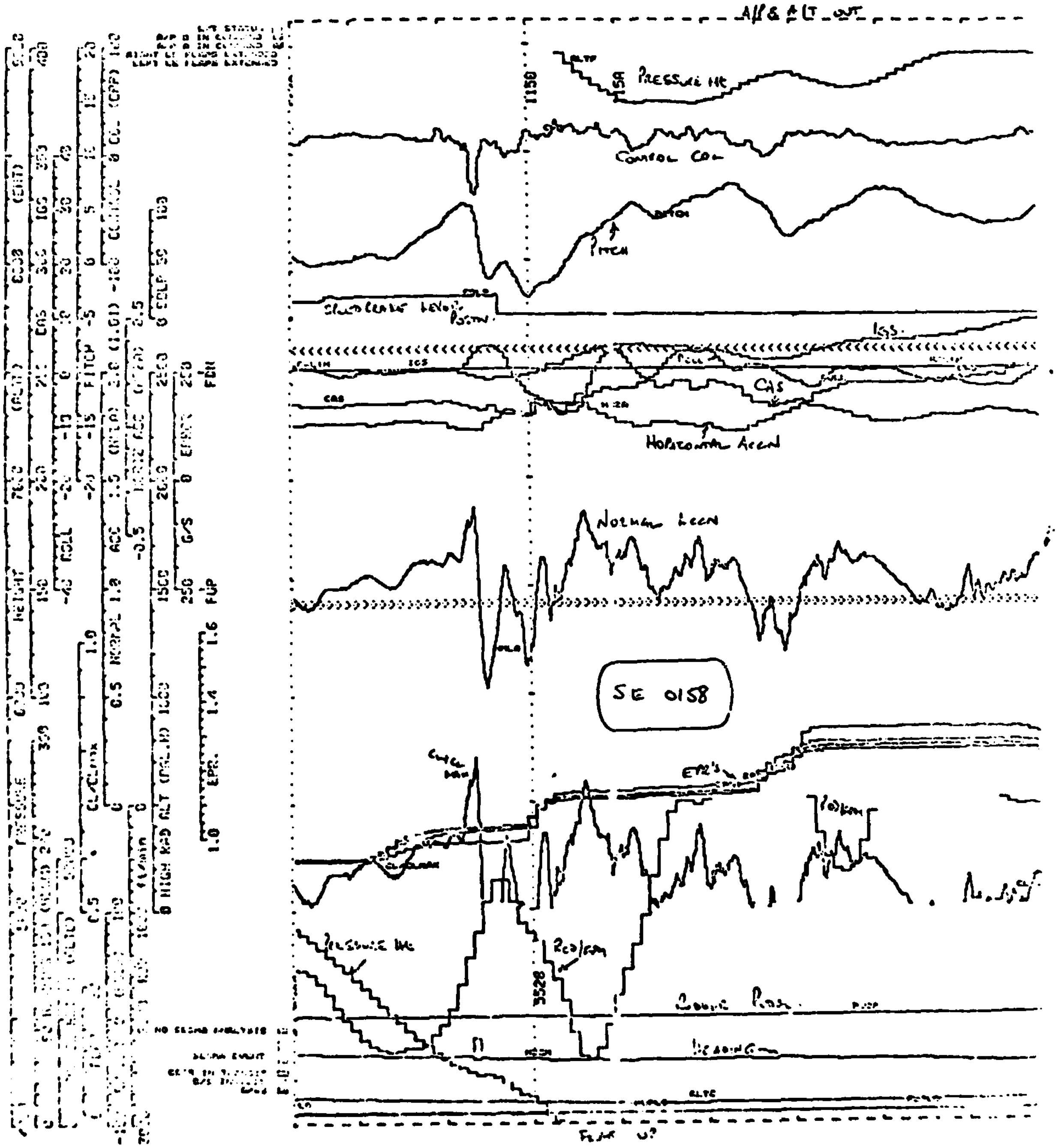
T/O ... R/W Alt TOP W/C VR V2 VNA B747-136 SIN-BLN JAN 1982 Land RUW R/W Alt W/C VAT
... 024 94 29 -19 150 159 169 264438 218 114 0 144

Stick-IT Event	Time	Length	Phase	Weight	Value	Detect level	Alert level	IAS	Alt	Mach	Flap	SAT	Wind	F	f
... STICK SHAKE	1149	0:01	4	276458	0	0	0	232	4228	0.381	0	19	001 45	7	22

First Event Lat 4 39 N Long 114 4 E Block ...

Alert		Circulate	Follow up			Airline nr:
✓						4965
Verified	Checked	Runway	T/L	D/L	M/T	CARDRP nr:
✓	LG LW	21				0158

289



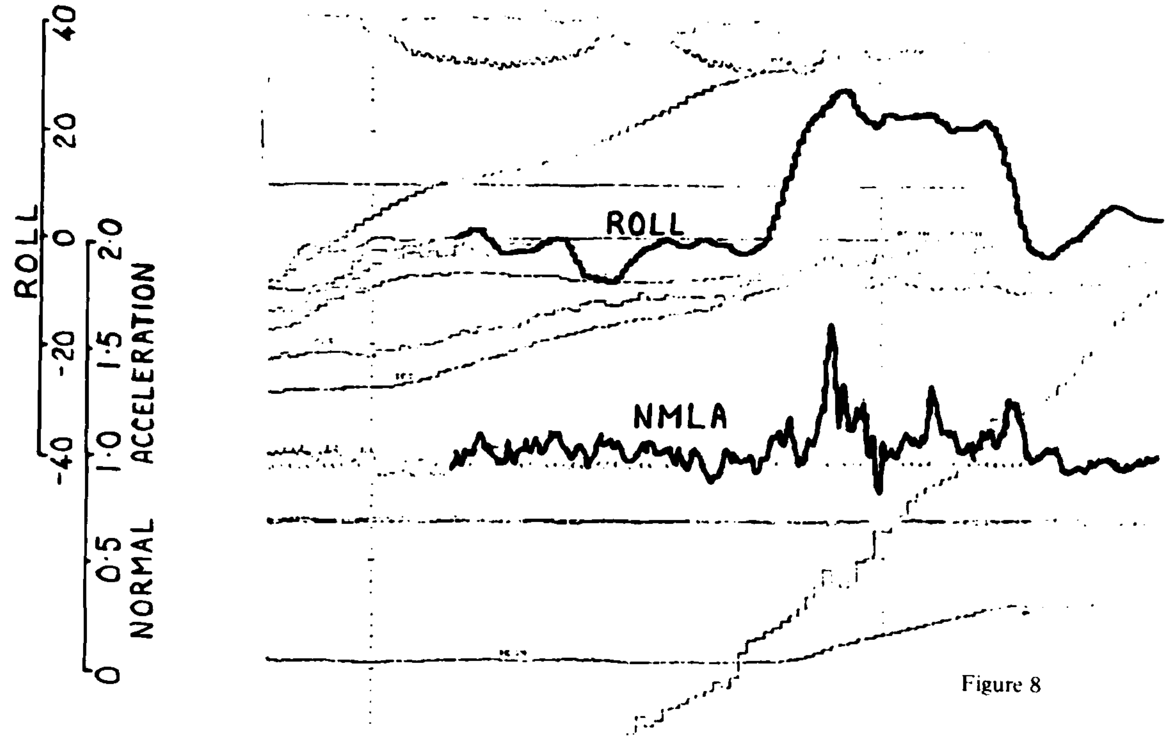


Figure 8

Figure 9

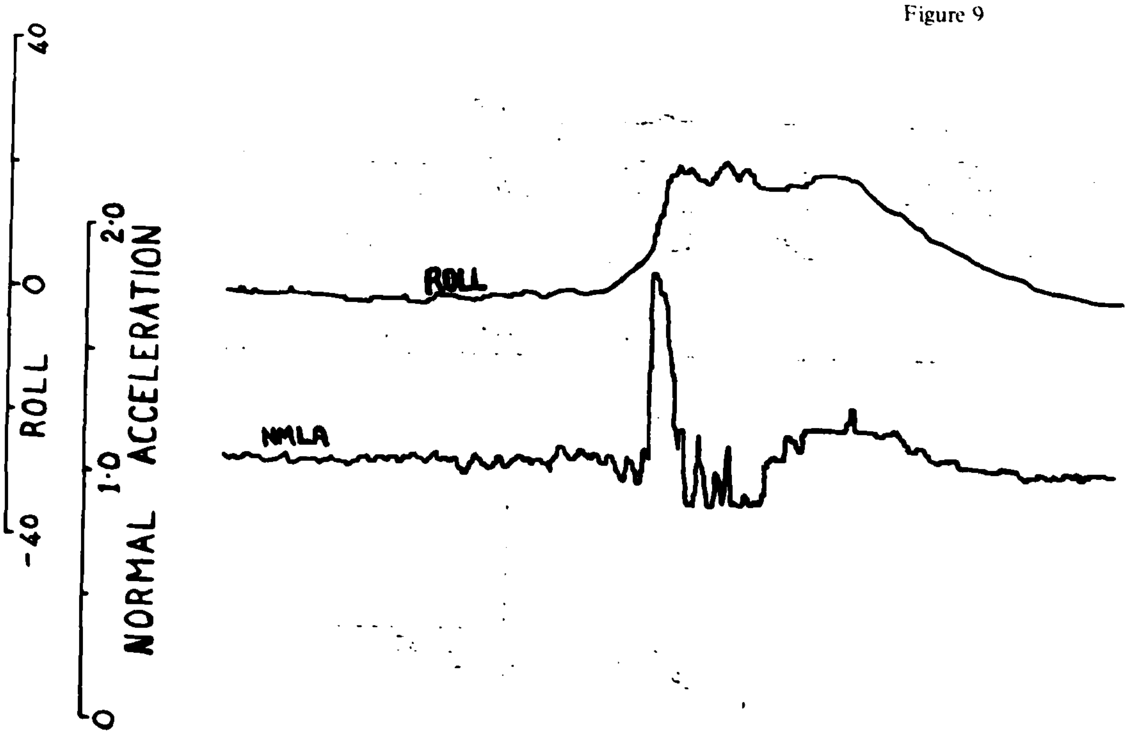


Figure 10

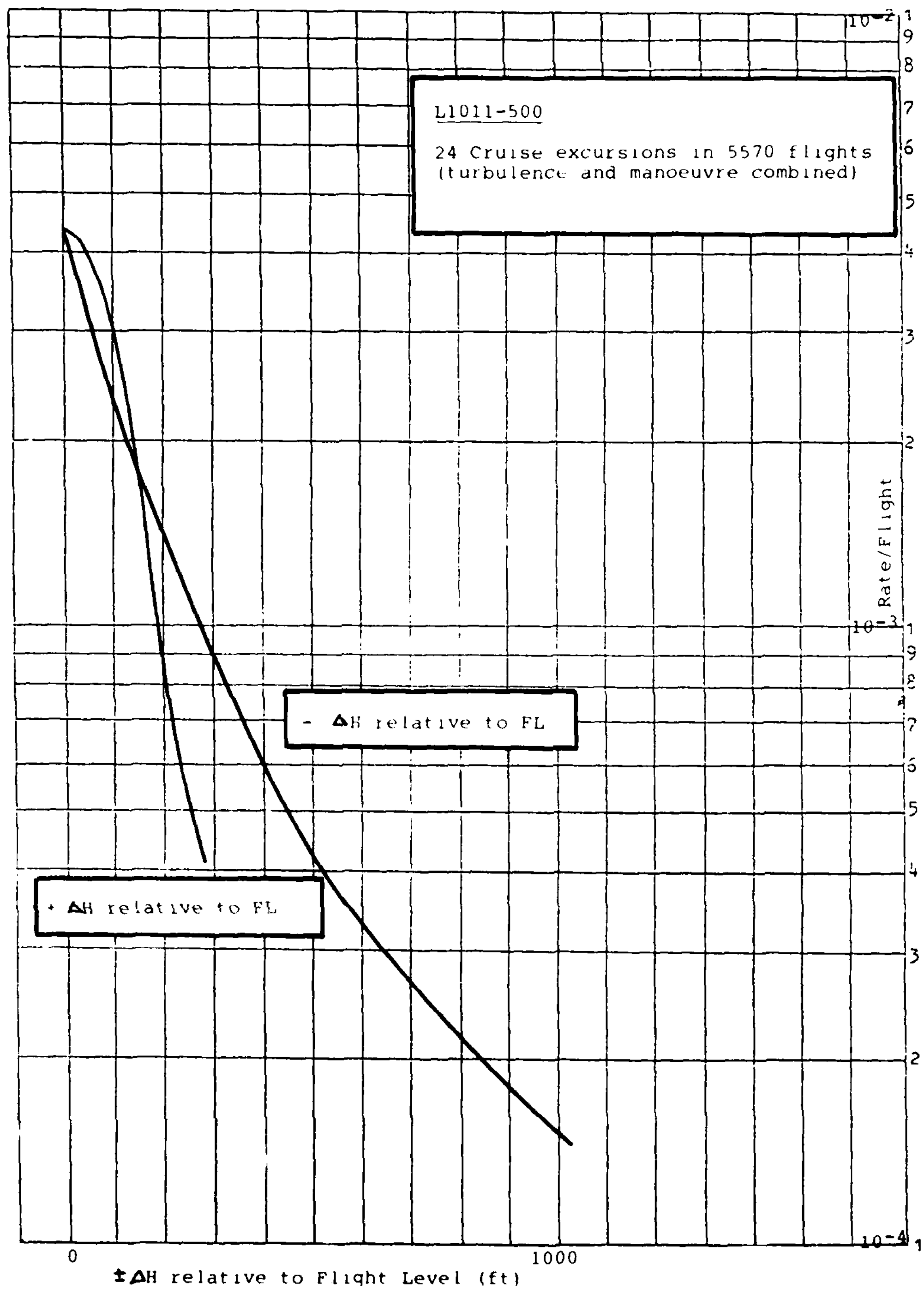

L1011-500
24 Cruise excursions in 5570 flights
(turbulence and manoeuvre combined)
- ΔH relative to FL
+ ΔH relative to FL
Rate/Flight
10^{-2}
10^{-3}
10^{-4}
0
1000
±ΔH relative to Flight Level (ft)

Figure 11

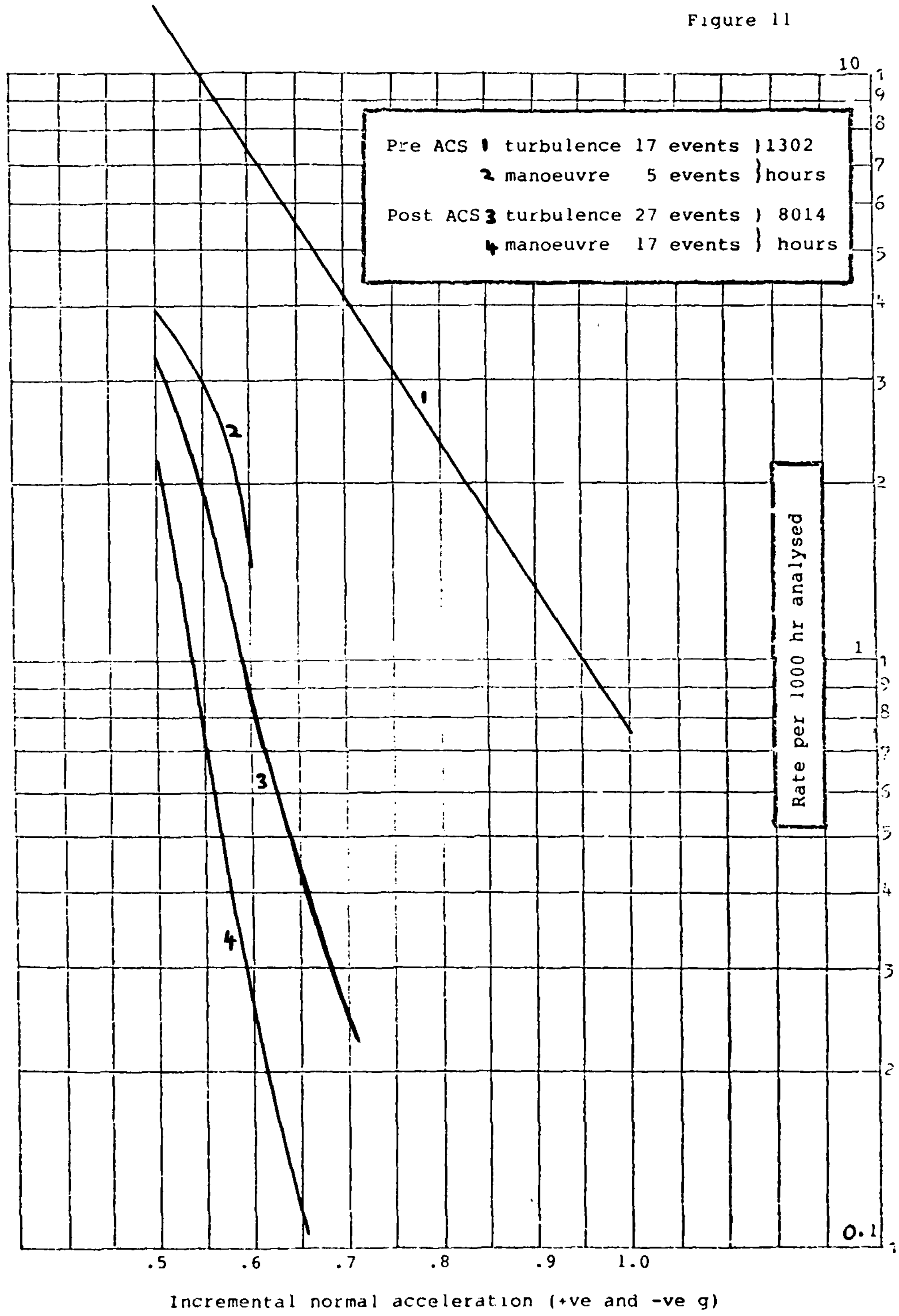
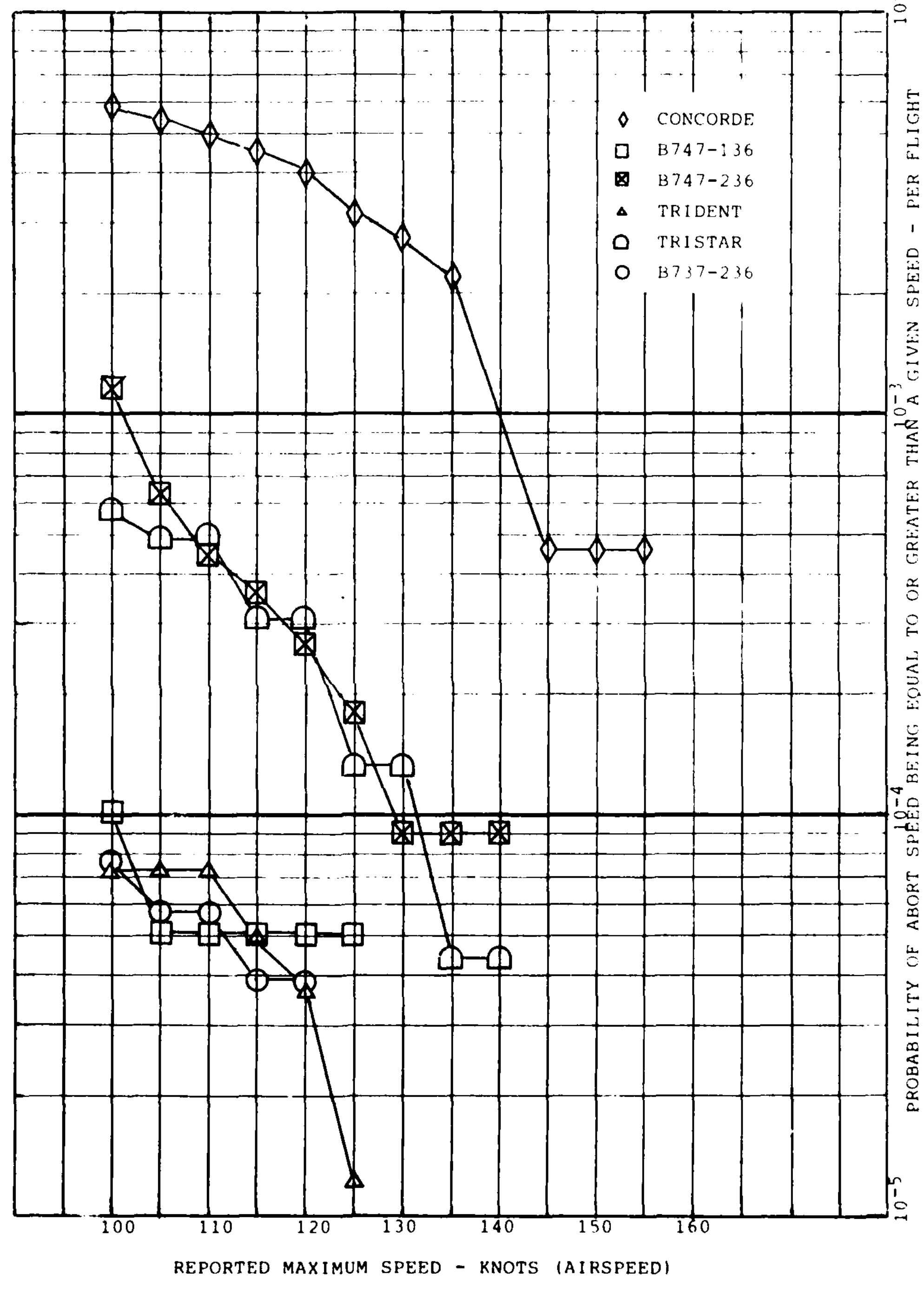

10
Pre ACS 1 turbulence 17 events } 1302 hours
2 manoeuvre 5 events
Post ACS 3 turbulence 27 events } 8014 hours
4 manoeuvre 17 events
1
2
3
4
Rate per 1000 hr analysed
1
0.1
.5
.6
.7
.8
.9
1.0
Incremental normal acceleration (+ve and -ve g)

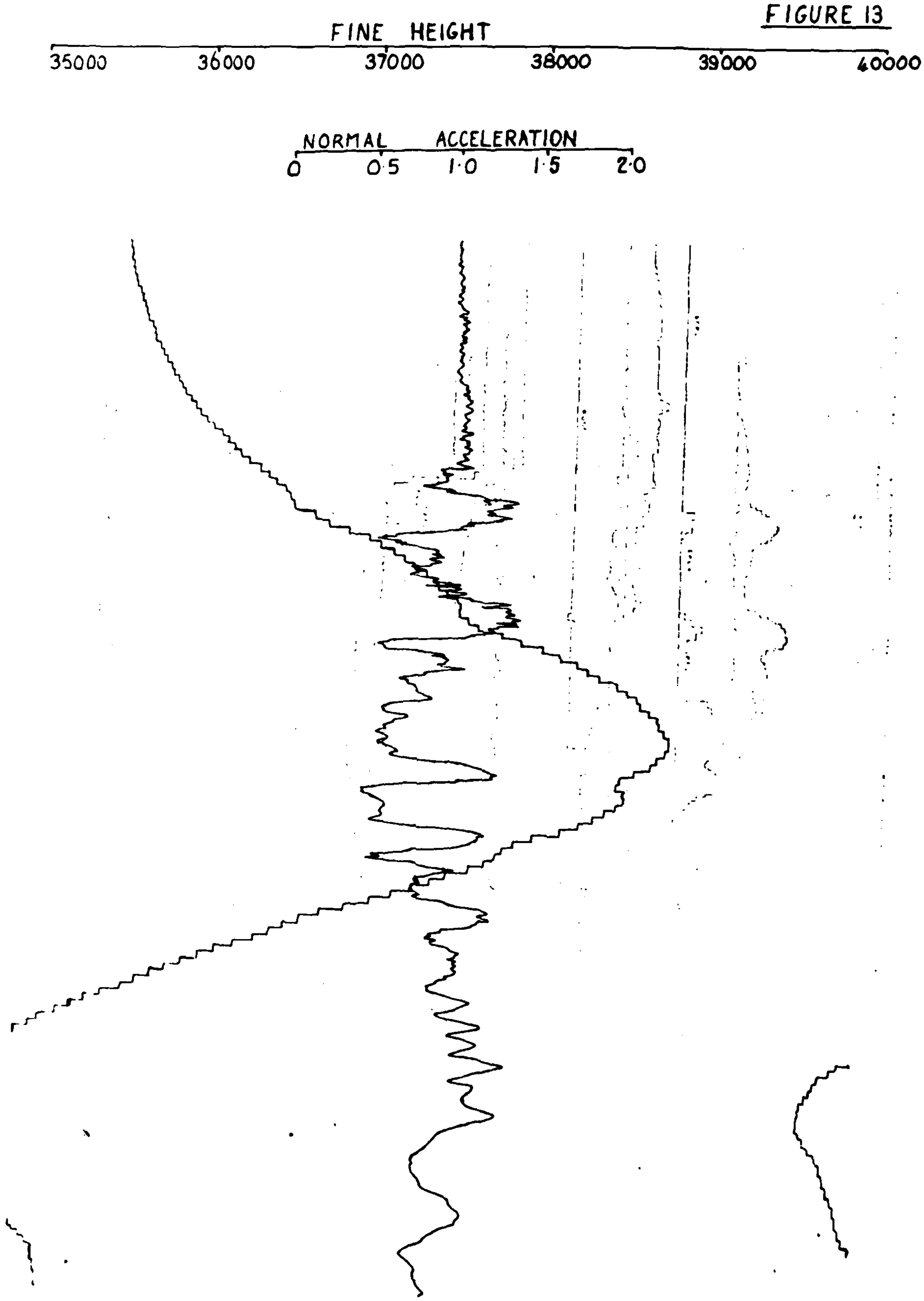
Figure 12

PROBABILITY OF AN ABORT FROM SPEEDS ABOVE 100 KNOTS
SPLIT BY AIRCRAFT TYPE



C

FIGURE 13


FINE HEIGHT
35000
36000
37000
38000
39000
40000
NORMAL ACCELERATION
0
0·5
1·0
1·5
2·0

TWO DECADES OF AIR CARRIER JET OPERATION

by
Edward C. Wood, Director, Engineering & Maintenance
George P. Bates (Consultant)
Flight Safety Foundation, Inc.
5510 Columbia Pike
Arlington, Virginia
22204
USA

SUMMARY

AGARD and the Flight Mechanics Panel, in holding this third conference on aircraft accidents, operations and safety have recognized the close relationship between civil and military aviation safety problems. This presentation examines the accident record of the world fleet of air carrier jet aircraft from 1960 to 1981. Comparisons of total accidents, fatal accidents, and hull losses for this time period are reviewed against numbers of departures and hours flown. It is shown that there is a close correlation between fatal accidents and hull losses and that there has been a general stabilizing of the accident rate in the past decade of two to four accidents (hull losses) per million departures with the probability that, if no changes are introduced, this rate will remain essentially unchanged.

Since hull losses reflect fatal accidents, a geographic comparison is made of these losses for 1975 and 1981 showing that, although significant improvements have been made in these countries, Asia, Africa and Central and South America remain the areas with the highest accident rates.

Costs of accidents are also presented from 1975 to 1981. Although the rates of accidents have remained essentially constant, the associated costs of those accidents has increased dramatically.

Phases of operations where accidents have occurred are also examined, showing that over 50 percent of the fatal accidents have occurred during 14 percent of the operational exposure time spent in initial approach, final approach, and landing. Of this 50 percent, nearly 40 percent have occurred in the initial and final approach phases.

The data tells us that our airframes systems and powerplants, as well as maintenance air traffic control and weather factors have reached a high level of reliability and that the human factor is still the prevalent factor in the existing accident rates. Operator error has remained at over 70 percent.

Since it is reasonably predictable that future accidents will occur in or near a major airport and will probably involve post crash fire, safety measures in this area are also discussed.

INTRODUCTION

In order to discuss accident and hull loss statistics, it is appropriate to review the definition of these terms. Figure 1 shows the U.S. National Transportation Safety Board terms which have been used in this presentation.

ACCIDENT DEFINITIONS

U. S. NATIONAL TRANSPORTATION SAFETY BOARD

"AIRCRAFT ACCIDENT" MEANS AN OCCURRENCE ASSOCIATED WITH THE OPERATION OF AN AIRCRAFT THAT TAKES PLACE BETWEEN THE TIME ANY PERSON BOARDS THE AIRCRAFT WITH THE INTENTION OF FLIGHT UNTIL SUCH TIME AS ALL SUCH PERSONS HAVE DISEMBARKED, IN WHICH ANY PERSON SUFFERS DEATH OR SERIOUS INJURY AS A RESULT OF BEING IN OR UPON THE AIRCRAFT OR BY DIRECT CONTACT WITH THE AIRCRAFT OR ANYTHING ATTACHED THERETO, OR THE AIRCRAFT RECEIVES SUBSTANTIAL DAMAGE.

"FATAL INJURY" MEANS ANY INJURY THAT RESULTS IN DEATH WITHIN 30 DAYS.

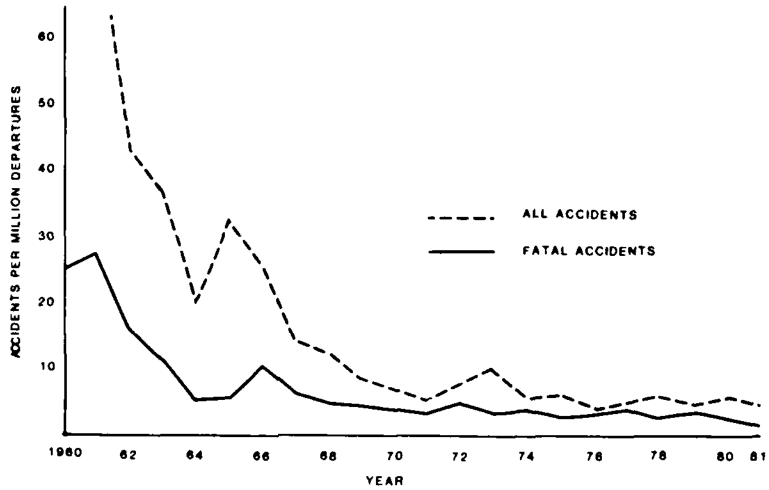
"HULL LOSS" MEANS DAMAGE DUE TO AN ACCIDENT THAT WAS TOO EXTENSIVE TO REPAIR, OR THAT, FOR ECONOMIC REASONS, THE AIRCRAFT WAS NOT REPAIRED AND RETURNED TO SERVICE.

FIGURE 1

The air carrier jet fleet has grown to a figure of over 6500 aircraft and has amassed nearly 189 million flight hours involving some 136 million departures. The accident rate has steadily decreased until the early 70's when a leveling of the rate began and has remained essentially the same since that time (Reference 1). A careful examination of the last decade is necessary to determine where the concentration of efforts must be applied in order to further reduce the present rate.

DISCUSSION

The world air carrier jet fleet total accident rates and fatal accident rates are shown in Figure 2.

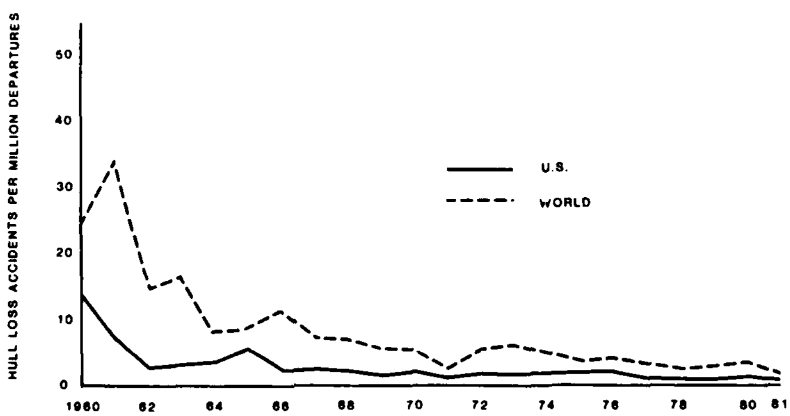


ACCIDENT RATE WORLD-WIDE JET FLEET

FIGURE 2.

This figure illustrates the significant improvement of accident rates since the early days of air carrier jet operation until the early 1970's when a lessening of this decreasing rate began and continues at the present time. For comparative purposes, plots of all accidents are shown along with those of fatal accidents. As would be expected, the trends are similar.

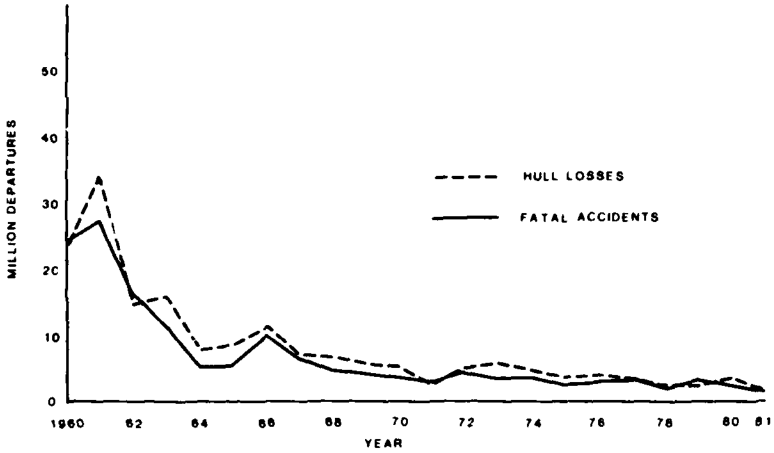
Hull losses for the same time period are shown in Figure 3.



HULL LOSS RATES AIR CARRIEP JET FLEET

FIGURE 3.

The similarity between fatal accident rates and hull loss rates is illustrated in Figure 4.

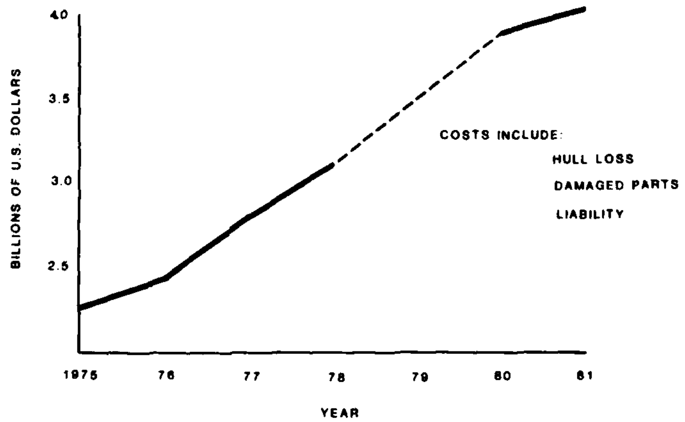


COMPARISON OF HULL LOSS & FATAL ACCIDENT RATE

FIGURE 4.

It is clearly shown that in today's air carrier jet fleet, a hull loss accident will probably be a fatal accident. The flattening of these rate curves could lead one to conclude that we are at a point where further significant gains will not be made and that continued expenditure of funds for safety improvement will produce little effect. To dispel this theory, we will examine the costs of accidents.

Hull losses can amount to 50 or 60 millions of U.S. dollars for a modern jet transport and, when combined with the expenses of damaged parts and liability costs, become an important consideration for an operator. In Figure 5 we see that in spite of the fatal accident/hull loss rate being stabilized at a fairly low level, the cost of an accident has continued to rise at an alarming rate, emphasizing the need for better safety management of today's sophisticated equipment (Reference 2).

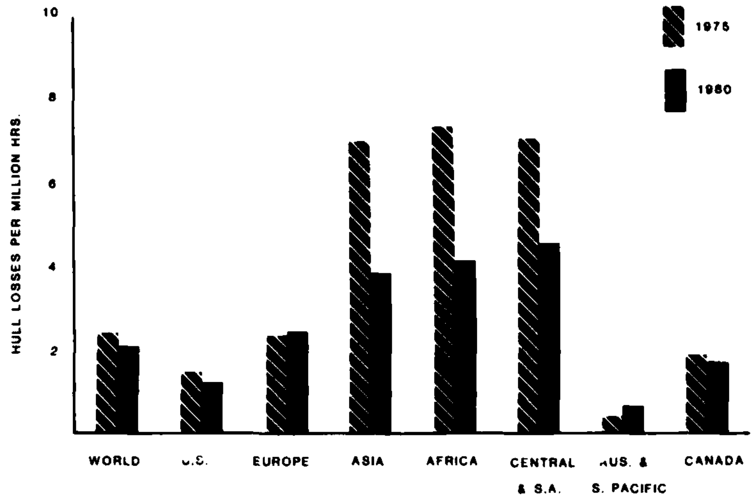


ESTIMATED ACCIDENT COSTS

WORLD JET FLEET

FIGURE 5.

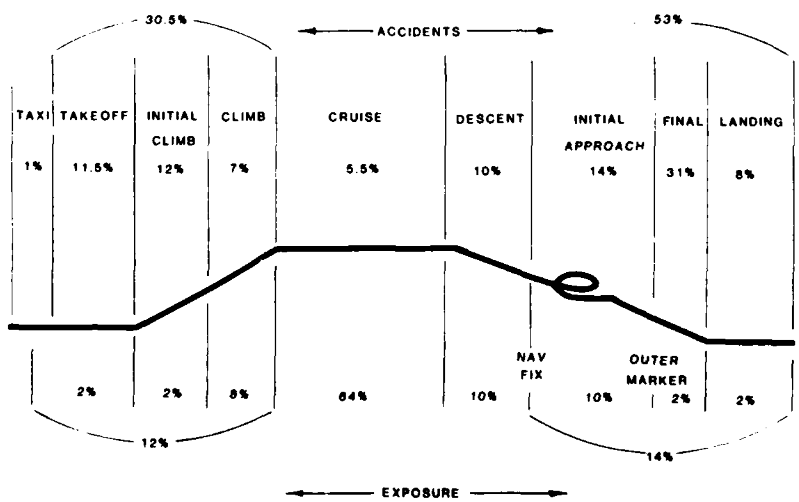
Continuing our overview of the air carrier jet accident history, Figure 6 is a geographic depiction of hull losses. The years 1975 and 1980 were chosen since they fall within the past ten year period of fairly constant accident rates. In general, all areas of operation reflect improvement with significant rate reductions in Asia, Africa and Central and South America although these three geographic areas continue to experience the highest hull loss rates.



AIR CARRIER HULL LOSS RATE

FIGURE [illegible].

To complete our statistical review, we will consider the phases of operation where these losses are occurring and the associated cause factors. In Figure 7 we see that for the 22 year time period over half the accidents have occurred in about 14 percent of the operational exposure time which is made up of initial approach, final approach and landing. It further shows that, of these accidents, nearly 40 percent occur in the final approach and landing phases which account for only 4 percent of the exposure time. This means that when the next accident occurs, it is probable that it will happen on or near a major airport and, according to U. S. National Transportation Safety Board statistics on U. S. jet air carriers, the probability of a post crash fire in what would otherwise be a survivable accident is also likely.



HULL LOSS ACCIDENTS 1960-1981

FIGURE 7.

Figure 7 also suggests that if further improvements are achieved in the latter phases of a flight, these same improvements are likely to affect the initial phases of flight (takeoff), initial climb and climb) which could improve the rates of these phases which account for about 30 percent of the accidents in 12 percent of the exposure time.

The final statistical chart, Figure 8, shows the causal factors involved in the 22 years of accident statistics. The percentages are approximate however they too have remained essentially the same for the past decade.

WORLD JET AIR CARRIER FLEET
1960 - 1981

CAUSE FACTORS IN FATAL ACCIDENTS

COCKPIT CREW	70 - 75%
AIRFRAME,POWERPLANTS,SYSTEMS	13%
MAINTENANCE	3%
WEATHER	5%
AIRPORT,ATC	4%

FIGURE 8

The human element, presented here as cockpit crew, has continued to be the predominant causal factor. The airframe, powerplants, systems and maintenance aspects have reached a much higher degree of reliability but of course must not be ignored where safety improvements are considered.

WHERE DO WE GO FROM HERE?

One of the difficulties in reviewing the voluminous data regarding the jet fleet accidents for over 20 years is synthesizing this information into a form on which positive actions can be taken. Analyzing the details of each accident, although important from the standpoint of providing an immediate remedial action for what went wrong, is not treated in this presentation. Rather, a search for a commonality in these accidents is attempted.

The stabilization of the accident rates and causal factors in the last ten years have shown that there is really nothing new in commercial jet accidents (Reference 3). In 1972 the U.S. NTSB conducted a special forum on approach and landing accident prevention at which some 35 papers were presented on this subject (Reference 4). In 1976 the Flight Safety Foundation devoted their entire conference to the subject at their 29th Annual International Air Safety Seminar in Anaheim, California. The problems were recognized early in the development of these trends therefore this presentation can only re-emphasize and discuss what has been done and what remains to be done.

SAFETY MANAGEMENT

The role of the manager in safety improvement cannot be overstressed. The cost of equipment and the related liabilities make it imperative that the aircraft and its operating environment be treated as a system with emphasis on maintaining high skill levels in every aspect which affects safe handling and operation. The designer also must assure that all equipment involved in this system reflect considerations of the human factor to prevent misuse and subsequent failure to produce the desired level of safety.

The need for safety management at all levels has been demonstrated vividly in recent years when, in Pakistan, a DC-10 which was receiving hangar maintenance, was destroyed by fire when cleaning solvent which was being used in the proximity of a titanium bolt grinding operation was ignited by sparks. Fire extinguishers were not immediately available and the hangar deluge system did not work. A DC-8 and a DC-9 were also consumed in separate occurrences by oxygen system fires during maintenance. These occurred in Europe and Canada respectively. Although hangar and ramp fires do not usually result in fatalities, the cost to the operator can be significant.

At the other end of the spectrum was the Turkish DC-10 which crashed in France in 1974 due to the improper latching of a cargo door. Three hundred forty-five occupants were killed making this one of the worst disasters in commercial aviation history.

COMMUNICATION

There are many accidents in which a causal factor was the lack of communication (Reference 5). Air New Zealand's Flight 901 is a classic example. This was the DC-10 which crashed on the slopes of Mt. Erebus in November 1979 killing all 257 occupants. Although the investigating board cited many factors which led to the catastrophe, the airline had changed the computerized flight track of the aircraft and had not notified the flight crew of this action.

The Trans World Airlines Boeing 727 which crashed on approach to Dulles Airport outside Washington D.C. involved as a causal factor the lack of proper communication between the cockpit and air traffic control, resulting in a premature descent. The 92 occupants were killed.

The prompt and accurate transfer of information extends beyond the cockpit. It involves all aspects associated with the operation of aircraft including maintenance, air traffic control and training of both flight crews and cabin attendants.

Communication within an aviation organization is certainly important but it is only one part of the effort. Information must be constantly transferred between organizations and between countries. This is particularly important to countries which have not yet amassed the experience which has been acquired by other operators. They should not have to learn by their own accidents what is already known to other operators.

This transfer of experience can be accomplished in many ways. The Aviation Safety Reporting System (ASRS) is one system utilized in the U.S. which accommodates the anonymous reporting of unsafe practices or occurrences. It is a system by which trend analyses can be conducted so that remedial actions can be taken and the system includes monthly newsletters in which selected reports are published for the information of those who are interested.

Seminars and conferences are another means of information transfer. This Flight Mechanics Panel of AGARD is a prime example as are the annual Safety Seminars sponsored by the Flight Safety Foundation. In order to continue, these types of meetings must receive domestic and international support.

WEATHER

Weather continues to be a causal factor in many of our accidents and in recent years considerable emphasis has been placed on the effects of wind shear. A three-year program has been initiated which will attempt to determine the efforts of low-level wind shear performance, test and evaluate wind shear detection and warning systems and will extend the usefulness of Doppler radars for detecting conditions of which pilots and the public need to be warned. This program is the Joint Airport Weather Studies Project (JAWS). It is a cooperative effort between the United States and Great Britain.

Also the Australian Department of Defence has recently completed a survey of 652 military and civilian Australian pilots and air traffic controllers to gather specific information on operational aspects of windshear (Reference 6). Information gathered through efforts such as these should assist in reducing accidents involving this phenomenon.

The importance of ground deicing was dramatically demonstrated in the 1982 Air Florida accident in the Potomac River in Washington D.C. Although many causal factors were involved, resulting in 11 reccommendations by the National Transportation Safety Board to the Federal Aviation Administration, the lack of standards for ground airframe deicing was of primary interest. This was another instance of an old problem not receiving adequate attention until after a major disaster.

GROUND PROXIMITY WARNING SYSTEMS (GPWS)

The extensive use of GPWS has been a major safety improvement in commercial aviation. Since 1975, when GPWS began to be extensively used, there has been approximately a 45% reduction in accidents involving controlled flight into terrain.

Presently, 21% or about 1300 of the world's commercial turboprop and jet aircraft have not been equipped with GPWS. Several countries including Canada, Finland, Greece and Spain do not require the use of these systems. Continued emphasis is necessary to encourage the installation and use of these systems.

ACCIDENT INVESTIGATION

After the fact information which is developed from accident/incident investigations is an absolute necessity to prevent future similar occurrences. The need for accuracy and thoroughness in these activities cannot be overemphasized and, again, the information gained must be promptly disseminated on a world-wide basis as promptly as possible.

The increased sophistication of the modern jet transport makes the investigation of these accidents more and more difficult without some previously recorded data such as that provided by Flight Data Recorders (FDR) and Cockpit Voice Recorders (CVR). Increasing the amount of data recorded could serve not only as an improved record of crash

information but could be used for maintenance trends and proficiency evaluations which provide operational cost savings and could be used to detect trends which could be useful in accident prevention.

TERMINAL AREA CLEANUP

Many of today's airports are still in need of improved runway approaches, overruns and shoulders which provide freedom from obstructions.

Accidents which have involved these factors might well have been incidents or minor accidents had major structural damage not been done by ditches, roadways or abrupt terrain elevations adjacent to the landing surface.

TRAINING AND CREW QUALIFICATIONS

Large quantities of information have been developed on this subject over the years, however, it must be continuously emphasized. Periodic review and updating of pilot rating requirements must be conducted. Psychological examinations for screening crew applicants must be put into effct as more information is gathered in this area. New techniques for training also must be put into use in the form of simulators and other computer techniques.

ANTI-MISTING KEROSENE (AMK)

Many landing accidents have resulted in fatalities when post-crash fires occur, preventing the rapid egress of the occupants. When there is no fire there is unlimited time for orderly egress when no injuries due to the crash itself have occurred (Reference 7).

The AMK program, which is being jointly pursued by the United States and Great Britain could provide a major break-through in preventing post-crash fires and fatalities. A full scale remotely controlled crash test involving a Boeing 707 is planned for this year in the Mojave Desert. The aviation community is anxiously awaiting the results of this test.

CONCLUSIONS

Although the jet air carrier accident rate has appeared to stabilize at a low level for the past decade, it must not be interpreted that there is nothing more to be done in safety in the operation of these aircraft.

- Even if the accident rate remains the same for the coming decade, the costs of these accidents will probably continue to increase dramatically.
- Techniques which have proven successful in achieving the present low accident must be continually emphasized and improved as technology permits. In addition, new safety technology must be applied as it is developed.
- Human factors, in all facets of aviation, continues to be a predominant aspect in accident prevention and in this regard, the prompt and accurate transfer of information is continually important.

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AD-P002 699

Incident Reporting--Its Role in Aviation Safety and the Acquisition of Human Error Data

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EXECUTIVE SUMMARY

The rationale for aviation incident reporting systems is presented and contrasted to some of the shortcomings of accident investigation procedures. The history of the United State's Aviation Safety Reporting System (ASRS) is outlined and the program's character explained. The planning elements that resulted in the ASRS program's voluntary, confidential, and non-punitive design are discussed. Immunity, from enforcement action and misuse of the volunteered data, is explained and evaluated. Report generation techniques and the ASRS data analysis process are described; in addition, examples of the ASRS program's output and accomplishments are detailed. Finally, the value of incident reporting for the acquisition of safety information, particularly human error data, is explored.

BACKGROUND

Man's intelligence and technology have worked toward a goal of greater system safety. As a consequence, the aviation industry has witnessed vastly improved aircraft, avionics, air traffic control equipment, and ground-based facilities. As the hazard profile of these "hardware" elements has diminished, the reality of the presence and magnitude of human error as a causal factor in accidents has been indelibly impressed on those agencies, organizations, and individuals vested with the responsibility for minimizing the risks associated with aviation operations.

Aircraft accidents are frequently caused or severely aggravated by human error. In tracing the chain of causation of these accidents, safety investigators and researchers have been generally effective in determining the "what" of the event, but they are not as effective in addressing the "why" of the event. The "why" of an event very often involves the human factors associated with that mishap. Unfortunately, the nature of accidents results in several significant problems in the investigation of human performance issues.

First, and fortunately, accidents are rare events; consequently, the investigator or researcher is provided with relatively few data points to study and from which to draw insight.

Second, it is an unfortunate truism that pilots are usually the first at the scene of aircraft accidents; they are often unable to be of much help after the smoke clears.

Third, in a litigious society, concepts of legal and financial liability effectively distort or hinder the investigation process. Even if a crewmember survives the accident, the potentially staggering consequences of an admission of error frequently thwart timely and accurate fact-finding.

Because of these problems the investigation of aircraft accidents is characterized by a "post hoc" reasoning process, often using incomplete facts, in an effort to determine a plausible chain of causation. The factual findings and determination of probable causes resulting from this reasoning process are subject to inaccuracies and may or may not yield reliable data useable in pursuit of enhanced system safety; this is particularly true with regard to human error data.

The three problems cited above are not present in a confidential, incident reporting system. Compared to accidents, incidents occur frequently. Pilots and crewmembers survive incidents, albeit sometimes with difficulty, and are characteristically willing to share the experience in the presence of assurances of protection. Finally, since the vast majority of incidents do not result in any liability the threat of adverse legal or financial consequences is usually not present.

Safety investigators have long been aware of, and concerned about, the monotonously repetitive pattern of human failure characteristic of accident statistics. This awareness and concern has been particularly well-developed within the aviation community. Many attempts have taken place over the past decades to obtain comprehensive information concerning operational problems in aviation, particularly in the case of those events that fortunately fell short of being accidents. One of the responses to this need for more and better information concerning operational and human error problems has been the implementation of incident reporting systems. The history of aviation incident reporting systems can be traced back to the early 1940's; the idea is not a new one. However, despite the anticipated potential of incident reporting programs, the desired productivity has not materialized; this is primarily due to the fact that many contributors were, and are, hesitant to describe their own errors and misconceptions, especially to an authority which has the power to discipline or economically penalize the reporter.

In recognition of the need for more and better information concerning operational and human problems in the United State's National Aviation System, the Federal Aviation Administration in May, 1975, implemented an Aviation Safety Reporting Program (ASRP), whose purpose was to improve the flow of information of possible significance to air safety investigations and research. To encourage the submission of reports, the agency offered a limited waiver of disciplinary action to those who provided timely information concerning incidents, and to others involved in those incidents, unless the occurrences involved a criminal offense, an aircraft accident, reckless operation, willful misconduct or gross negligence.

Notwithstanding the FAA's promise that information reported under the ASRP program would not be used against the provider of the data, it soon became obvious that there were misgivings in the aviation community regarding the regulatory and enforcement agency's role in the collection and use of the often sensitive incident data. For that reason, the FAA asked the National Aeronautics and Space Administration (NASA), an independent research organization without regulatory or enforcement authority, to act as a "third party" in the program. NASA was requested to design a modified incident reporting program and to take over responsibility for the receipt, processing, analysis and deidentification of those aviation incident reports. In addition to its ability to fulfill the role of a disinterested intermediary, NASA saw a unique benefit to its research capability deriving from ongoing access to the human factors data generated by an incident reporting system. NASA accepted the FAA's proposal; the Aviation Safety Reporting System (ASRS) began operations on April 15, 1976.

CHARACTERISTICS OF THE SYSTEM

The designers of the ASRS had as their objective an incident reporting system that would have the characteristics of utilization, utility, and effectiveness. Essentially, the mission was to design and operate a system that the community trusted and with whom it would communicate; in addition that system had to be capable of constructively using the data received, and the program's product had to be reflected in effective applications within the aviation system.

The objectives of NASA's ASRS program are as follows:

- To make available a confidential reporting system which can be used by any person in the national aviation system.

- To operate a computer-based system for storage and retrieval of processed data.

- To provide an interactive analytical system for routine and special studies of the data.

- To maintain a responsive system for communication of data and analyses to those responsible for aviation safety.

The basic statement of purpose of the program can be found in the ASRS Memorandum of Agreement between NASA and the FAA ... "This system will be designed primarily to provide information to the FAA and the aviation community to assist the FAA in reaching its goal of eliminating unsafe conditions and preventing avoidable accidents". More specifically, the purpose of the ASRS program has been defined as:

> Identifying deficiencies and discrepancies in the national aviation system to provide a knowledgeable basis for improving the current aviation system; and providing data for planning and improvements to future systems.

In an age of information and communication, the acquisition of safety data regarding aviation incidents should not present a significant challenge to program planners and managers; and in fact, most members of the aviation community have historically exhibited a willingness to share information, especially about accidents,

hardware, and other partys' actions. However, the mission of the Aviation Safety Reporting System is to obtain incident data provided by the participants in those events; more specifically, the ASRS database is designed to reflect the participant's assessment of the situation or occurrence and his or her role in that condition. Other information systems exist to assemble descriptive, statistical, or second and third-party data; but the ASRS mandate involves the task of gathering analytical first-party data, especially information that addresses the "why" of an event as reflected in the behavior exhibited by the participants.

The first step toward satisfaction of the ASRS mandate was to design a system in which the aviation community, both individually and collectively, could place a high degree of trust; furthermore, that trust from the community needed to be matched by consistent credibility on the part of the ASRS program and the program's management. It was decided that the elements of trust and credibility could be best served by an incident reporting system that was voluntary and promised total confidentiality.

While mandatory reporting systems may produce greater quantities of data, the quality of data from such systems may suffer from superficiality and doubt on the part of the report source as to its ultimate use. Voluntary information systems, on the other hand, have usually been characterized by higher quality reporting from individuals motivated by a genuine desire to see an issue pursued beyond the "filling-in-the-blanks" phase of safety investigation. By providing the motivated volunteer with the equally important assurance of absolute confidentiality, the ASRS design accommodates both the researcher's need for quality, comprehensive data and the reporter's desire for selectivity and anonymity.

One of the major attributes of the ASRS program is its cooperative nature. Since the inception of the ASRS, a vital degree of oversight and guidance has been provided by virtually every segment of the aviation community. The result of this cooperation is the existence of an incident reporting system that is viewed as a safety information resource to be utilized by all elements of the aviation community.

Of equal importance to the elements of voluntariness and confidentiality is the lack of an enforcement mandate in the charter of the organization vested with the responsibility for the incident reporting program administration, data analysis and information management. This consideration made the selection of the National Aeronautics and Space Administration a logical one in the search for a disinterested third-party. NASA's role as third-party intermediary between the members of the aviation community and the Federal Aviation Administration has often been characterized as that of an "honest broker" attending to the best interests of both sides.

A collateral issue to the design of a system which encouraged voluntary incident reporting was that of immunity for those individuals electing to report to the ASRS. The issue of immunity is bifurcated. Immunity protection can apply to the use of the data obtained, in which case the issue is termed "use immunity"; by the same token, immunity protection can refer to the shielding that the individual obtains from disciplinary action in exchange for his or her information; this is referred to as "transactional immunity". In conjunction with the NASA pledge of confidentiality for report sources, the FAA offered both forms of immunity to contributors to the ASRS program. The first, use immunity, was established in the form of promises contained in the FAA Advisory Circular and the FAA/NASA Memorandum of Agreement which set forth that "... FAA will not seek and NASA will not release to the FAA any information that might reveal the identity of [persons filing reports and persons named in those reports]". The concept of use immunity was further strengthened in 1979 with the implementation of Federal Aviation Regulation #91.57 which states:

> The Administrator of the FAA will not use reports submitted to the National Aeronautics and Space Administration under the Aviation Safety Reporting Program (or information derived therefrom) in any enforcement action, except information concerning criminal offenses or accidents which are wholly excluded from the Program.

To a large degree use immunity and confidentiality are intertwined; in the context of the ASRS program neither of these two basic elements has been altered or even challenged by any party to the system.

From the beginning of the ASRS program in April of 1976, the issue of reporter protection from enforcement actions, transactional immunity or the "waiver of disciplinary action", has been a point of contention. Although not specifically requested by the aviation community in the 1975-1976 period, the waiver of disciplinary action was offered by the FAA as an element of the ASRS concept.

It should be noted at this point that the waiver of disciplinary action associated with ASRS incident reporting has always been viewed by NASA as an issue between the FAA and the aviation community. While recognizing it as an element of the overall ASRS concept, NASA, which has no authority to pursue enforcement actions or grant any immunity from them, has essentially taken an observer position on the issue of transactional immunity.

The original waiver of disciplinary action that accompanied the ASRS in April of 1976 remained in force until July 1, 1979. Following a period of controversy over the need for the existence of transactional immunity for ASRS reporters, the waiver of disciplinary action was modified to reflect the current provisions as set forth in FAA Advisory Circular 00-46B:

> The filing of a report with NASA concerning an incident or occurrence involving a violation of the Act or the Federal Aviation Regulations is considered by the FAA to be indicative of a constructive attitude. Such an attitude will tend to prevent future violations. Accordingly, although a finding of a violation may be made, neither a civil penalty nor certificate suspension will be imposed if:
>
> (1) The violation was inadvertent and not deliberate;
>
> (2) The violation did not involve a criminal offense, or accident, or action under Section 609 of the Act which discloses a lack of qualification or competency, which are wholly excluded from this policy;
>
> (3) The person has not been found in any prior FAA enforcement action to have committed a violation since the initiation of the ASRP of the Federal Aviation Act or of any regulation promulgated under that Act; and
>
> (4) The person proves that, within 10 days after the violation, he or she completed and delivered or mailed a written report of the incident or occurrence to NASA under ASRS.

Transactional and use immunities have become a primary consideration in the ASRS concept. It is conceivable that a successful incident reporting system could be launched without transactional immunity, but use immunity is essential.

Reports containing information relating to aviation accidents (as defined by National Transportation Safety Board Regulation 830.2) and criminal activities (as codified in Title 18 of the U.S. Code, Annotated) are exempt from both the immunity and confidentiality provisions of the ASRS program. Because the ASRS and its staff members can not be above the law in the sense of withholding accident or criminal information, all such information is forwarded to the appropriate investigatory bodies and not retained in the ASRS database. It should be noted that the individuals who have submitted the reports of accidents or criminal activity are notified after the data's receipt of the requirement placed on the ASRS to forward the information to the proper federal agency; this courtesy is extended primarily to let the person know what happened to the data; it is also done to explain the loss of immunity and confidentiality.

Two coincidental, but different categories of motivation prompt contributors to the ASRS program to report their experience. The first category, direct personal advantage through confidentiality and immunity, has already been discussed. The second, enhanced system safety, is a product of what the ASRS staff does with the data that has been volunteered. In essence this issue simply requires ASRS to recognize that it must achieve and feed-back program results, otherwise the majority of data submitters will stop seeing value in program participation and not report their experiences.

Feedback to the aviation community can be both direct and indirect. The most immediate response to the reporter community is the direct feed-back provided to the reporter following submission of an ASRS report. Few frustrations match that of voluntarily submitting data derived from personal experience to a governmental body which has been requesting such data, and then not having that contribution acknowledged. Immediately upon deidentification of each ASRS report form the individual who submitted that report is provided, by return mail, with the following:

- The Identification Strip section of the ASRS report form, date-stamped and bearing the internal tracking number for that I.D. strip; in addition, where possible, ASRS analysts are encouraged to add a short, personal note to each I.D. strip from reports they have worked;

- Two blank ASRS Reporting Forms to replace the one submitted to ASRS;

- A letter of appreciation to the reporter for his or her contribution to the ASRS program;

- A copy of the current issue of Callback, the ASRS's informal safety publication. This enclosure not only passes on safety information, it also exhibits the ASRS's capability for constructive data usage and timely dissemination of contributed data.

This direct return response is accomplished within days, usually no more than four or five, of the date of receipt of the report at the ASRS offices. Not only has the reporter received the I.D. strip for immunity purposes, he or she is also made immediately aware of the report's receipt, data usage, and acknowledgement of the government's appreciation for his or her efforts and concern in pursuit of enhanced aviation safety.

The indirect feed-back to the reporter community takes the form of evidence of data usage through Alert Bulletins, periodic technical reports, the ASRS's monthly safety publication Callback, and awareness of the community's ability to access the ASRS database for legitimate safety investigations and research. In other words, the individual reporters, and their professional organizations or trade associations, are made aware of the fact that useful information is coming out the production end of the ASRS process in a timely fashion.

THE REPORT ANALYSIS PROCESS

In the seven years of its existence, the ASRS has received more than 34,000 reports; human errors can be found, and confessed to, in more than 70 percent of these reports. Most reporters are frank to admit to their own mistakes, and will go into detail in describing the circumstances, character, and outcome of the incident. Contributors to the ASRS seem to genuinely care about their role in an event and take pains to report actions, emotions, and perceptions accurately in the face of often critical circumstances. We have been impressed with the care and effort put into the writing of most of the reports while at the same time we have lamented the lack of detail in a few others, some of which described possibly serious potential problems. We have thought that many reports were probably trivial in terms of any impact on safety - until later, when it became clear that reports which appear trivial in isolation can help to point to an underlying factor of real importance. The ASRS staff is frank to admit that we are unable to characterize a "trivial" report, for in concert with other reports, it may assist in understanding a genuine problem. Many reports stand as monuments to the dedication of their authors to aviation safety; some of the best are reprinted in full, after deidentification, in ASRS program reports.

ASRS reports are received and processed daily by members of the ASRS staff. The reports are read by an attorney who has a background in aviation law and aviation safety. If they involve criminal acts, they are transmitted in identified form to the U.S. Department of Justice for investigation. If they refer to an aircraft accident, they are forwarded, identified, to the National Transportation Safety Board. If, in the NASA reviewer's opinion, they contain time-critical safety data, they are singled out for priority handling. The reports are then forwarded by bonded courier to the ASRS program contractor's office for analysis.

The ASRS program managers made an early decision to use the services of a civilian contractor to assist in the design and management of the ASRS program. Following the usual competitive procurement process, Battelle Memorial Institute's Columbus Laboratories was selected as the ASRS program contractor. Battelle has established a base of operations for ASRS activity adjacent to NASA's Ames Research Center, thereby allowing for quick and easy communications between NASA's ASRS management and the program's contractor, who is now responsible for the majority of ASRS report processing and database research.

The reports are given to analysts, each of whom is an expert in the area of air traffic control, general aviation or air carrier operations. The analysts study the reports, and decide whether further direct contact with the reporters is necessary or desirable. If so, they initiate such "callbacks" by telephone. Callbacks have the unique advantage of increasing the rapport between reporter and analyst, while at the same time enhancing the post-event learning and analysis process for both parties. Thereafter, or if further contact is not desired, they remove the identification strips; the strips are logged out by serial number and returned to reporters with a new report form.

By the time the reports have passed through the initial analysis phase, all identifying information contained in the original report has been removed. This deidentification process includes the removal and return of the reporter's identification strip; in addition, any individual or company name, as well as any aircraft names or numbers, are deleted from the report and appropriate substitutions entered in their place.

Analysts study the information provided by reporters either in their reports or in subsequent contacts. If they believe that a report contains time-critical safety data, it becomes the basis for dissemination of an Alert Bulletin. They may augment the information provided in the report by constructing or adding charts or other graphic material. They are not permitted to verify or refute the information provided by a reporter through contact with other persons; the ASRS mandate prohibits this, and the System's resources do not permit investigation of reported incidents.

Analysts then add to the reporter's narrative a synopsis, their analytic comments and informal notes. *It should be noted* that when ASRS reports are analyzed and evaluated an attempt is made to discern both human and system factors associated with the reported event or situation. It is often impossible to attribute cause and effect relationships to such factors; although it is usually possible to categorize various factors as having an "enabling" or "associated" relationship to the chain of causation. In addition to coding these enabling and associated factors, the analysts also determine and code the "recovery" factors involved in the event, thereby recording their analysis of the reasons this event was an incident and not an accident. The analysts code the report to incorporate data describing the attributes of the occurrence; descriptive and diagnostic terms are added as a final step. The entire package is then checked by a second reviewer-analyst for completeness and prepared for computer entry.

Typists transfer each report package to magnetic tape; the tapes are copied and sent to a computer facility where they are read onto disk files for storage. The information management system which houses the ASRS data is Battelle's Automated Search Information System (BASIS). BASIS is a very effective, flexible analytic tool for large bodies of free text and coded data. Report narratives, synopses and analyses are entered in the computer in free text format. A substantial number of coded entries describing each report is also available; these entries, along with the descriptors and diagnostic terms, are indexed and are therefore readily available as search terms.

SYSTEM OUTPUT AND ACCOMPLISHMENTS

The ASRS program's output has two basic functions. The first is to notify the FAA and the aviation community of the existence of alleged hazards in the system. The second function is to attempt to provide an explanation for the presence of hazard conditions; essentially an attempt to achieve an understanding of the "why" of certain conditions or situations.

The ASRS is capable of disseminating data in several ways. The output of the program to date consists of:

- More than 770 Alert Bulletins - time-critical notices from the ASRS to persons or organizations in a position to effectively investigate, and possibly cure, an alleged hazard reported to the ASRS.

- 464 special data requests - responsive database searches requested of the ASRS by the FAA, NTSB, and other members of the aviation community. Requested studies vary from the simple to the extremely complex; responsive searches have ranged from a few database statistics to lengthy studies.

- 14 Program Reports - periodic publications containing samples of constructive deidentified reports received by the ASRS, selected Alert Bulletins and the responses to them, and one or more reports on research studies performed by the project staff. The fifteenth program report is currently in preparation.

- 11 Research Reports and Technical Papers - single topic research reports dealing with aviation safety problems; addressing primarily human factors issues, these reports have dealt with subjects such as fatigue, information transfer problems, controlled flight toward terrain, cockpit distraction, and altitude deviations.

- 47 monthly safety publications - Callback, an easy-to-read one-page newsletter designed for the light, but timely, expression of safety issues relevant to the entire aviation community.

The products of ASRS data usage are distributed to the aviation community by several means order to publicize the uses and value of the incident reporting system. ASRS Program Reports are supplied to over 40,000 individuals through company or organizational distribution channels, direct mailings from a list maintained by the ASRS staff, and through the National Technical Information Service. Technical and Contractor Reports are distributed by direct mail from a recipient list created and maintained by the ASRS staff. The Callback publication is provided to any member of the aviation community who has expressed to the ASRS office a desire to be placed on the mailing list for that publication. Finally, special requests for deidentified information from the ASRS database are available to the aviation community for legitimate safety investigations, research, and training activities.

In addition to the ASRS products cited above, the incident reporting system has been a key source of data for several aviation safety review bodies which have assisted in the formulation of national aviation policy. Extensive incident information was supplied from the ASRS database for:

- The President's Task Force on Aircraft Crew Complement

- The National Institutes of Health Select Panel on Mandatory Pilot Retirement

- The Flight Safety Foundation's Air Traffic Control Evaluation Task Force

- The Air Traffic Control Association Review and Analysis of Air Traffic Control Terminal Area Operations

- The National Transportation Safety Board's Review of the U.S. Air Traffic Control System

Non-documentable and intangible contributions to safety constitute a second class of program accomplishments from the ASRS. ASRS has significantly improved communication among the various segments of the aviation community, including FAA, DOD, NTSB, and NASA. All elements of the community have worked together on the system; all have used its data in the pursuit of solutions to safety problems. The common database has made it possible to reach consensus on some issues; in other cases, it has permitted more rational and focussed advocacy by the proponents of differing points of view. In several cases involving national aviation policy, ASRS has been virtually the only source of incident, as opposed to accident, data. There is no other similar database, and there is considerable doubt whether one could be accumulated under different ground rules.

While there appears to be no effective means of measuring the impact of ASRS data in the field of aviation education and training, it is known that there has been widespread use of ASRS material by flight instructors, flight schools and air carrier training facilities, as well as military training and safety organizations. The contents of the ASRS program's research publications and Callback, the monthly safety bulletin, are frequently reproduced in airline, flightcrew, and military safety education publications.

Among the subtleties of the impact of ASRS activities on system safety is that of moral suasion for the purpose of leverage. It is not an uncommon occurrence for a request to be made of the ASRS staff to provide data or a publication to support a legitimate safety improvement that is on the verge of acceptance but needs a little extra push. Because of the depth of information in the ASRS database, particularly human error data, and because of the program's credibility within the aviation community, ASRS alert bulletins and research data have been used to achieve safety objectives in need of impartial support. This use of ASRS product has been evident in actions and communications instituted by elements of the community, the military, and governmental agencies.

One of the important benefits of incident reporting to a program like ASRS takes place before the report ever reaches the program office. Program participants have expressed the notion that the act of having to organize and express the relevant facts and issues associated with a given event or situation has proven to be an extremely valuable learning experience for the reporter. Because of the program's assurances of confidentiality, reporters have often gone beyond a basic recitation of the facts to probe their own motivations, misconceptions, proficiency, and other considerations that may have contributed to the the factors that made up the incident. The event analysis and performance critique that takes place at both ends of the reporting process is clearly a significant, but unmeasureable, benefit of the ASRS program.

The most obvious, as well as the most undocumentable, category of ASRS achievements is the element of accidents avoided and deaths prevented. It is impossible to document a non-event. However, given the array of research, alert bulletins, publications and assistance offered and utilized as a result of ASRS operations, it seems reasonable to assert that the presence and products of the ASRS have prevented accidents and saved lives.

Summary

When you want to know more about an occurrence, or why a person did what they did in the course of events, the best approach seems to be to simply ask the participants. First-hand experiental input is not a foolproof method of data acquisition. It is subject to the biases and fears of the reporter; but it is usually better than interrogation of witnesses and non-participants, or second-guessing. It is our experience that a voluntary, confidential, non-punitive incident reporting system is a logical and effective means of acquiring unique data, as well as supplementing data generated by conventional accident investigation techniques and other system

monitoring programs. A properly-structured incident reporting system's great advantage is that it has the strength and the means to ask, and frequently answer, the question "why?" whenever one is confronted with a "what". There is no substitute for knowing why a system failed or a human erred. If we understand why things happen, we may be able to prevent them from happening again or at least protect the participants, or the system, from the consequences of subsequent events. The potential for constructive uses of incident data seems to be especially promising in the field of human behavior; incident reporting is a tool which permits the cooperative examination of human behavior in complex systems, using data supplied directly by the participants in that system. ASRS activities and research have been oriented toward issues associated with the role of the human in the operational aspects of aviation. The program's structure and principles have permitted the ASRS to compile an extensive body of incident data; more specifically, the assurances of confidentiality and the availability of transactional immunity have resulted in the creation of a large and comprehensive human factors database for use by aviation investigators, researchers, planners and policy-makers throughout the world.

The success of the Aviation Safety Reporting System in the United States, and the international emergence of programs similar to ASRS, support the position that incident reporting in other nations and other disciplines can be effective in achieving a better understanding of system deficiencies and human error.

L'ANALYSE DES ENREGISTREMENTS DE PARAMETRES OUTIL INDISPENSABLE A LA SURVEILLANCE ET AU CONTROLE DE L'EXPLOITATION

par

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I. INTRODUCTION

L'analyse systématique des vols par lecture des enregistrements de paramètres est pratiquée à la Compagnie AIR FRANCE depuis près de dix ans.

Elle constitue l'une des bases de la Politique de Prévention des Accidents dont il nous a paru utile de rappeler les concepts fondamentaux.

Après un bref historique, seront rapidement exposés les moyens mis en place, les methodes et, surtout, la "philosophie" qui a conduit à l'établissement d'un "Protocole" dont l'application rigoureuse a permis d'obtenir l'adhésion et la participation des équipages.

Seront ensuite exposés les principes de fonctionnement

- de la "Commission d'Analyse des Vols" chargée d'examiner les dossiers les plus significatifs
- de la "communication anonyme" destinée à recueillir auprès de l'équipage, sans qu'il soit nécessaire d'identifier celui-ci, les informations complémentaires utiles à la Commission.

Enfin, seront présentés quelques uns des principaux résultats obtenus grâce à l'Analyse systématique des Vols.

II. POLITIQUE DE PRÉVENTION DES ACCIDENTS - CONCEPTS FONDAMENTAUX

2.1. - Interactions Homme/Machine Environnement

a) Les incidents ou anomalies de vol sont généralement causés par une défaillance de l'Homme, de la Machine ou par un facteur défavorable de l'Environnement.

b) L'accident est rarement le résultat d'une simple cause. Il est généralement le résultat de la conjonction de facteurs appartenant à plusieurs, voire à l'ensemble, de ces principaux domaines qui constituent une chaîne d'événements.

c) Par suite de l'évolution rapide de la technologie, l'interface Homme/Machine mérite une attention particulière.

2.2. - Nature des accidents

a) Chacun des facteurs ayant contribué à un accident a généralement déjà été identifié dans des incidents précédents, isolément ou dans une combinaison différente.

b) Chaque facteur peut avoir, dans la chaîne des événements, un rôle déterminant dans la gravité du résultat final. Il suffit, en effet, de la présence d'un seul d'entre eux pour faire la différence entre l'incident et l'accident.

2.3. - Relation entre nombre d'incidents et probabilité d'accident

Il y a relation entre le nombre d'incidents et la probabilité d'accident, ainsi qu'entre la fréquence des différents types d'événements et leur gravité.

2.4. - Intérêt de chercher à diminuer le nombre d'incidents même mineurs et d'étendre le domaine d'investigation à toute anomalie.

Ces relations entre nombre d'incidents et probabilité d'accident peuvent être illustrées schématiquement par une pyramide dont la base serait constituée par les incidents mineurs et le sommet par l'accident.
Si on admet par exemple, que pour 300 incidents il y a une probabilité de 30 incidents majeurs pouvant eux-mêmes conduire à un risque sérieux d'accident, le fait de diminuer le nombre des incidents réduira la probabilité d'occurrence des incidents majeurs et, de là, le risque potentiel d'accident.

Le principe fondamental de la Prévention des Accidents repose donc sur l'investigation approfondie de tout incident mais également sur l'élargissement du domaine d'investigation à toute anomalie survenue en vol pour pouvoir réduire la base de la "pyramide".

2.5. Boucle de prévention des accidents

Le premier stade de la Prévention des Accidents est l'établissement de normes de certification des avions, la définition d'une réglementation de l'exploitation et du niveau des licences équipage.

Ceci est effectué en fonction de certaines hypothèses, donc a priori. Il convient de certifier ces hypothèses dans les conditions de réalisation des vols, c'est-à-dire en fonction de l'environnement technologique, opérationnel et humain et, surtout, en fonction de l'évolution de celui-ci.

C'est l'objectif du Contrôle et de la Surveillance de l'exploitation qui permet de détecter et d'évaluer tout incident, anomalie ou difficulté et, par son action a posteriori, de fermer la boucle de Prévention des accidents tant à l'intérieur de la Compagnie elle-même que bien en amont, auprès du constructeur, voire des administrations chargées d'établir les normes de certification ou de définir la réglementation.

Nous verrons qu'en ce domaine, l'analyse systématique des enregistrements de paramètres de vol peut jouer un rôle fondamental.

III. COMPARAISON DES DEUX PRINCIPALES SOURCES D'INFORMATION

3.1. Rapport équipage

Le principal moyen d'avoir connaissance des incidents ou anomalies en vol était, jusqu'alors, le rapport des équipages.

Les incidents liés à une panne ou anomalie de fonctionnement du matériel sont bien connus car faisant l'objet de comptes rendus des mécaniciens navigants établis après chaque vol et exploités par les services d'entretien. Ce systeme fonctionne bien.

Par contre, on constate que les incidents de caractère opérationnel qui impliquent davantage des hommes facilement identifiables – équipages ou contrôleurs de la circulation aérienne – ne sont pas toujours connus.

Les raisons en sont multiples :

- Tout d'abord, l'équipage peut avoir ignoré l'incident ; par exemple s'il survole le relief avec des moyens insuffisants en vol aux instruments, ou s'il est victime d'une illusion d'optique en vol à vue.

- Il peut également arriver qu'il sous-estime l'importance de l'anomalie ou l'intérêt de la rapporter parce qu'il la considère comme un cas isolé.

- Il peut aussi considérer que son rapport risque de ne pas être exploité.

- Il peut craindre de se trouver impliqué, voir sa réputation engagée. Bien des gens pensent qu'il est difficile de reconnaître des erreurs, même vis-à-vis de soi. Il leur est donc encore plus difficile de rapporter des erreurs à la compagnie qui les emploie, ou l'administration qui leur délivre la licence.

- Il peut craindre, enfin, de mettre en cause d'autres professionnels de l'aviation.

Il existe un moyen susceptible de procurer des informations extrêmement intéressantes : il s'agit des enregistreurs de paramètres de vol installés à bord des avions pour faciliter les enquêtes en cas d'accident.

3.2. Enregistrements de paramètres

Ceux qui équipent les avions modernes permettent de reconstituer les trajectoires, de connaître la configuration avion, le déroulement du vol, ainsi que les principaux paramètres de fonctionnement des moteurs.

Leur utilisation systématique par les compagnies se heurte à deux obstacles : le coût de l'opération et les réticences des équipages.

Or, certains résultats de cette utilisation intéressent l'entretien des avions et de leurs équipements, et ceci peut contribuer à rentabiliser les investissements nécessaires en matériel et en personnel. En effet, la possibilité de connaître de nombreux paramètres, en particulier ceux concernant les moteurs, et le fait que les relevés de vol établis par les mécaniciens navigants peuvent manquer de précision en cas de variations rapides ou être incomplets car il est difficile de noter plusieurs paramètres en même temps, a suscité un très grand intérêt de la part des services d'entretien pour l'utilisation des enregistreurs de vol.

Reste la réticence des équipages. Celle-ci, qui repose principalement sur la crainte de sanctions, se manifeste dans toutes les compagnies de tous les pays, ce qui explique qu'à ce jour "l'analyse des vols" à l'aide des enregistrements est assez peu pratiquée. Une grande compagnie aérienne a tenté, il y a quelques années, de faire admettre par l'administration de son pays et par ses équipages la création d'un tel système, ce qui aurait permis en particulier la suppression d'un des deux contrôles annuels en vol réglementaires pour les pilotes, mais l'entreprise a buté sur la difficulté de déterminer les tolérances de pilotage acceptées par tous et a dû être abandonnée.

Dans quelques compagnies européennes il est cependant possible de pratiquer officiellement l'analyse des vols. C'est le cas de la compagnie AIR FRANCE qui utilise systématiquement depuis près de dix ans, avec le plein accord des représentants des Organisations Professionnelles du Personnel Navigant Technique, les enregistreurs de vol pour obtenir des informations sur les conditions dans lesquelles se déroule l'exploitation, dans le cadre d'un protocole qui en définit les modalités de fonctionnement.

IV. PROTOCOLE ENREGISTREUR

Les principes fondamentaux de ce protocole, signé le 24 avril 1974, sont les suivants :

1. - l'analyse systématique des enregistrements doit se faire de façon strictement anonyme
2. - elle ne doit pas être utilisée pour l'évaluation des performances individuelles
3. - elle ne peut conduire à la prise de sanctions à l'encontre des équipages (si l'anomalie n'a pas entraîné d'incident ou n'a pas été révélée par une autre source)
4. - participation des représentants des Organisations Professionnelles aux travaux d'analyse des vols au sein d'une Commission.

Les anomalies (cas d'analyse des vols) ne sont plus examinées par un Chef Pilote, mais par une Commission regroupant un grand nombre de cadres PNT et dont nous examinerons la composition plus loin.

V. HISTORIQUE DE "L'ANALYSE DES VOLS" ET MOYENS MIS EN PLACE

Jusqu'en 1974, les enregistreurs des paramètres de vol utilisaient, pour la plupart, des supports photographiques permettant l'enregistrement d'un nombre réduit de paramètres (10 paramètres).
L'analyse des enregistrements se limitait alors aux incidents et se faisait, en ce qui concerne le domaine opérationnel, directement par la Division de Vol par examen visuel.
Ce type d'analyse se trouvait forcément limité. Il a été exploité pour les Caravelle, B.727 et B.707.

L'importance du réseau AIR FRANCE (300 à 400 étapes de vol par jour) et la décision de procéder à l'analyse systématique de tous les vols ont nécessité la mise en place de moyens relativement importants en matériel et en personnel.

Il fut procédé, en 1974, à la création d'un Service centralisé, indépendant des Divisions de Vol, rattaché directement au Directeur des Opérations Aériennes, et placé sous l'autorité de l'Officier de Sécurité des Vols.
Cet organisme, appelé Service Sécurité et Analyse des Vols, est chargé de promouvoir une politique de prévention des accidents, l'analyse systématique des informations enregistrées étant un des principaux moyens mis à sa disposition.

L'obligation faite par l'Arrêté du 28 mai 1975 d'équiper les avions certifiés après le 30 novembre 1969 d'un enregistreur magnétique d'accident, capable d'enregistrer un nombre beaucoup plus élevé de paramètres, a permis d'envisager une utilisation beaucoup plus élargie.

Outre sa capacité plus importante (60 à 90 paramètres) l'intérêt d'un enregistrement magnétique digital est qu'il peut être soumis directement à un traitement par ordinateur après lecture. Il a été décidé de procéder à l'installation d'un deuxième enregistreur à cassette d'une autonomie d'enregistrement de 50 heures appelé PMR (Performance Maintenance Recorder), l'enregistreur d'accident n'étant pas adapté de par son autonomie et les difficultés d'accès à l'enregistrement (incorporé et à boucle fermée).
Il a donc été conçu un système capable de détecter automatiquement certains écarts par rapport à un profil de vol standard, ou d'alerter l'exploitant sur la possibilité d'une anomalie quelconque.

Un premier système appelé SYDAS a été mis en place en 1976. Sa capacité est maintenant insuffisante. Grâce à l'expérience acquise par l'utilisation de ce premier système, il a été possible de réaliser un système plus complet, plus élaboré et plus performant : le système CARINE "Chaîne d'Analyse et de Restitution des Informations Enregistrées" qui sera mis en exploitation courant 1983.

VI. STRUCTURE DE LA CHAÎNE DE TRAITEMENT DES ENREGISTREURS

Le système CARINE comprend, essentiellement :

- une unité centrale composée
  - . d'un mini-ordinateur CII Honeywell Bull - modèle 92 de la gamme Mini 6 d'une capacité mémoire de 1 Mo,
  - . d'une mémoire tambour de 512 Mo,
  - . d'une unité de mini-disques de 8 Mo,
  - . d'une console de commande,

- un ensemble d'entrée permettant
  - . la lecture des cassettes PMR (Performance Maintenance Recorder) et
  - . la lecture des enregistreurs d'accident DFDR (Digital Flight Data Recorder) au travers d'un synchronisateur,
  - . la saisie des données d'identification des étapes par liaison IBM ;

- un ensemble de sortie comprenant
  - . une unité d'enregistrement sur bande magnétique pour les statistiques,
  - . une imprimante,
  - . trois écrans graphiques,
  - . une table traçante.

VII. DÉROULEMENT DU TRAITEMENT

Conçu comme un outil d'aide aux personnels chargés de l'analyse des vols (exploitants), le système CARINE effectue un traitement que l'on peut schématiquement décomposer en quatre phases :

- une phase automatique,
- une phase manuelle,
- une phase dite "conversationnelle",
- une phase finale de constitution du dossier dénommé "Cas d'analyse des Vols",

et qui permet d'assurer les principales fonctions suivantes :

1. En PHASE AUTOMATIQUE :

   . lecture et traduction des informations enregistrées (cassette PMR),
   . identification des étapes,
   . classement des étapes selon la classification :

   1 : étapes sans anomalie,
   2 : étapes avec anomalie(s) probable(s),
   3 : étapes avec anomalie(s) certaine(s),

   . stockage sur mini-disque des étapes de classe 2 et classe 3 pour traitement ultérieur,
   . relevé et stockage des données pour traitements statistiques,
   . édition d'un état dit "Etat d'anomalies".

2. En PHASE MANUELLE, l'exploitant intervient pour

   . examiner l'Etat d'anomalies,
   . évaluer les anomalies en fonction de l'environnement opérationnel afin de juger de la nature et de l'importance des anomalies.

A ce stade, l'exploitant juge de la poursuite ou non de l'analyse de chaque cas détecté en phase automatique.

3. En PHASE CONVERSATIONNELLE, l'exploitant procède à une analyse plus approfondie à l'aide de l'écran graphique qui lui permet de visualiser l'ensemble de la phase de vol concernée (paramètres, trajectoires) pour déterminer

   - le degré de gravité de l'anomalie,
   - le choix, la disposition, l'échelle des courbes de paramètres, des trajectoires et des profils destinés à la constitution du dossier.

   Cet écran permet également d'apporter des renseignements supplémentaires et d'éliminer des éléments superflus.

4. En PHASE FINALE de constitution du dossier d'ANALYSE DES VOLS

   l'exploitant procède à l'impression sur papier de l'ensemble des informations qu'il a sélectées sur l'écran graphique et constitue le dossier dit "Cas d'analyse des vols" qui comprend :

   - le dossier des données opérationnelles,
   - des extraits des manuels d'Exploitation et d'Utilisation rappelant les procédures et les consignes concernées,
   - un état de paramètres,
   - éventuellement, une trajectoire et un profil de vol.

VIII. EXPLOITATION DU DOSSIER "CAS D'ANALYSE DES VOLS"

8.1. Examen par l'Officier de Sécurité des Vols

Les dossiers de cas d'Analyse des vols sont soumis régulièrement à l'Officier de Sécurité des Vols qui les examine et retient ceux qui feront l'objet d'une présentation en Commission d'Analyse des Vols – les autres étant conservés à des fins statistiques ou adressés directement, pour information, à l'équipage concerné.
Il peut, en vue de compléter certains dossiers, décider d'interroger les équipages concernés par le système de Communication Anonyme afin d'obtenir des renseignements complémentaires sur les circonstances et éventuellement, les causes auxquelles les équipages attribuent les anomalies relevées par l'analyse des vols.

8.2. La Communication Anonyme

Destinée à compléter un dossier d'analyse des vols, la demande de renseignements auprès de l'équipage concerné se doit de respecter le principe d'anonymat du Protocole.

Dans ce but, le dossier, dont les paramètres d'identification du vol non nécessaires à la compréhension du cas sont supprimés, est reproduit en trois exemplaires (1 exemplaire pour chacun des membres de l'équipage technique) insérés dans une enveloppe cachetée adressée à

Monsieur le Commandant
du vol ....(n° du vol)
du ....(date)

Cette enveloppe est transmise au Chef du Service Technique du Centre de Vol concerné qui, après consultation du planning, la placera dans le casier du Commandant ayant effectué le vol.
Cette procédure respecte donc strictement l'anonymat ; en effet :

- l'Officier de Sécurité des Vols connaît les faits, mais ne connaît pas l'identité de l'équipage,
- le Chef du Service Technique connaît le nom du Commandant, mais ignore le contenu de l'enveloppe.

Il appartient au Commandant, s'il le juge utile, de faire parvenir un exemplaire du dossier à chacun des autres membres de l'équipage pour information demande de renseignement ou d'avis. Il pourra faire parvenir sa réponse, verbale ou écrite, sous forme anonyme ou non, à l'Officier de Sécurité des Vols, à son Chef Pilote, ou aux représentants mandatés des Organisations Professionnelles.

8.3. La Commission d'Analyse des Vols

Les dossiers de Cas d'Analyse des Vols — éventuellement complétés par les réponses aux communications anonymes — sont périodiquement (tous les 2 mois) examinés par la Commission d'Analyse des Vols.

a) Composition de la Commission :

Présidée par l'Officier de Sécurité des Vols, la Commission regroupe des représentants

- des Divisions de vol,
- des Services Techniques,
- des Divisions d'Instruction,
- du Service Niveau Professionnel,
- des Organisations Professionnelles Pilotes et Mécaniciens Navigants.

b) Travaux de la Commission :

Chaque dossier — précédemment transmis aux participants — fait l'objet d'un exposé suivi d'une discussion où chacun des membres de la Commission d'Analyse des Vols apporte son point de vue sur les causes possibles de l'anomalie, son importance, les enseignements à en tirer, les actions correctives à envisager.
Il peut être décidé de demander des renseignements complémentaires aux équipages ou d'entreprendre des études particulières en vue d'analyses ultérieures plus approfondies.

En fin de travaux, la Commission d'Analyse des Vols, pour chacun des cas étudiés, dépose des conclusions, formule des recommandations et propose des mesures correctives.

IX. LES RÉSULTATS DE L'ANALYSE DES VOLS PRATIQUÉE À AIR FRANCE

Telle que pratiquée depuis près de 10 ans dans le cadre du Protocole, l'Analyse des Vols a permis, outre de nombreux résultats dans le domaine de la sécurité des vols, de créer un climat de confiance parmi le Personnel Navigant Technique : on a constaté, d'une part, une augmentation du nombre de rapports équipages signalant des anomalies ou incidents de vol et, d'autre part, une demande de plus en plus fréquente des Commandants de Bord pour prendre connaissance du dépouillement de l'enregistrement de l'un de leurs vols afin d'apprécier le déroulement de telle ou telle manoeuvre ou procédure.

Nous citerons, enfin, certains résultats notables de l'Analyse des Vols :

9.1. Principales difficultés ou anomalies mises en évidence :

- risques résultant d'approches sur forte pente ou en forte régression de vitesse, assorties parfois d'atterrissages durs avec rebonds ;
- difficultés d'accrochage à haute altitude au voisinage du plafond (Boeing 747)
- risques résultant du non respect des minima (difficultés de stabilisation en conditions marginales) ;
- risques résultant de l'inobservation des procédures ou des règles relatives des tâches à bord (calage des altimètres, lecture de la radio-sonde, surveillance des paramètres de vol par le pilote non aux commandes, etc...).
- réduction de marges de survol des obstacles
- vitesses en dessous des vitesses requises en approche intermédiaire, pilote automatique, ou système automanettes engagé.

9.2. Principales actions correctives menées :

- création d'approches-types à effectuer en vol VMC (Conditions météorologiques de vol à vue) sur certaines pistes d'accès difficile, telles que la piste 32 à Marseille et la piste 21 à Ajaccio ;
- création ou simplification de procédures d'approches aux instruments, par exemple à Pointe-à-Pitre (piste 29) et à Tanger ;
- amélioration de la documentation concernant notamment la présentation des informations sur les feuilles de procédure. Par exemple, figuration d'aérodromes ayant donné lieu à confusion tels que les terrains civils ou militaires voisins de ceux de Porto ou d'Hanovre ; mentions concernant les procédures particulières de réglage des altimètres sur les aérodromes des Pays de l'Est à la suite d'erreurs de calage altimétrique ;
- mise au point et rappel de règles à appliquer en cas d'incapacité, complète ou subtile, du pilote aux commandes ;
- réexamen des conditions d'exécution, sous les aspects réglementaires, organisation et répartition des tâches à bord, des percées radio-dirigées et des approches indirectes ;
- diffusion d'un livret sur le vol en haute altitude.

X. BULLETIN D'ANALYSE DES VOLS RÉSERVÉ AU PNT

Les cas d'analyse des vols les plus significatifs font l'objet d'une diffusion périodique réservée au PNT : le Bulletin d'Analyse des Vols.

Regroupés par familles (Approches précipitées, Erreurs de position en approche, Non respect des procédures et minima, ...) ces cas sont présentés en reprenant, pour chacun, l'essentiel des éléments du dossier (exposé, trajectoire, profil, paramètres), le déroulement des travaux de la Commission, ses conclusions, recommandations et propositions de mesures correctives.

XI. STATISTIQUES

De 1978 à 1982, plus de 328 000 vols sur SE 210, B.727, B.707, ont été analysés : 4772 fiches d'anomalies (soit 1,45 %) ont été établies.

Ces anomalies ont été classées par types d'anomalies, types d'avion, jour ou nuit, etc.

Il apparaît que c'est en cours d'approche que la fréquence des anomalies est la plus élevée, notamment du fait de vitesse ou de pente de descente élevée en approche intermédiaire.

Une autre classification par région ou aéroport a été également effectuée. Celle-ci montre que sur certains aéroports tels que Ajaccio ou Marseille la fréquence des anomalies est beaucoup plus élevée que la moyenne.

XII. CONCLUSIONS

L'analyse des vols est devenue la base de la politique de prévention des accidents à Air France.

Près de dix ans d'expérience ont permis de définir des méthodes et de développer des moyens capables de couvrir les besoins pour l'analyse systématique de tous les vols.
La participation effective des équipages, grâce au respect du Protocole, est un des acquits les plus importants de l'Analyse des Vols.

Celle-ci, en fournissant une grande quantité d'informations, a permis de dégager des tendances et d'en identifier les causes. Des actions correctives ont été menées dans les principaux domaines d'action suivants : l'instruction, les procédures opérationnelles, la réglementation, la documentation, l'information et la sensibilisation des équipages. Elle conduit également à des interventions auprès des aéroports et des Centres de Contrôle de circulation aérienne.

L'Analyse des Vols est donc un maillon essentiel de la boucle de prévention des accidents.

FLIGHT PARAMETERS RECORDING FOR SAFETY MONITORING AND INVESTIGATIONS

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SUMMARY

Up to now the in service usage monitoring of the military aircrafts has been conducted mainly relying on g-counters installed on each aircraft, but the extreme complexity of the present combat aircrafts, the variety of their tasks and usage in service, suggest now the opportunity to have a continous monitoring of the structure and engine usage for checking the integrity and the life consumption.

An accurate and complete monitoring system, which is considered cost effective, presents, among the many advantages, those of allowing a greater flight safety. The system in development adopts, as basic philosophy, an on board recording facility on each individual aircraft of all relevant flight and aircraft parameters and a ground equipment performing two levels of analyis: (a) trouble shooting for structure, engine and systems limits exceedances or malfunctions after each flight during the normal turn around time giving a GO/NO GO indication for the next flight and (b) a detailed calculation of fatigue loading experienced during each flight and the consequent residual life evaluation for the critical items, thus allowing the detailed monitoring of the usage of each aircraft.

This system is based on the experience gained on the operations of the Italian combat aircraft and represents a new approach to the problem to guarantee the airframe and engine safety.

The effectiveness and the configuration of this Maintenance Recording system has been studied with the Italian Airforce which is going to implement the system in their Tornado operational line.

1. NECESSITY OF A MONITORING SYSTEM

The extreme complexity of a modern multi role aircraft as the TORNADO, the great variety of tasks it has to performe with the consequent very high number of stores and store combinations to be flown, the different in service usage of the aircraft, have led to consider highly desirable a continous monitoring of the aircraft operations to check the structural integrity and the life consumption.

The main advantages of such a monitoring, if performed accurately and extensively, can be identified as:

- greater flight safety
- optimization of aircrafts' usage
- proper planning of aircrafts' inspections.

In fact, the consideration of the theoretical and then very severe fatigue spectra used to design the structure (including the engine structures), drastically controls the in service life of the aircraft and requires, for each component of the fleet, a very heavy schedule of inspections related to flown hours and not to really experienced damage; on the contrary, the knowledge of the service usage of the aircraft, in peace time surely lighter, can lead to a relaxation of the inspections schedule with the same safety degree and high effectiveness.

To meet the above target, two main requirements have been put for the study of a monitoring system:

- fatigue life consumption calculation
- facility to check the possible exceedances of the most significant flight parameters (during each flight) in order to give a GO/NO GO indication for the next one.

Airframe, engine and some among the most important aircraft systems, are subject to the monitor to guarantee a complete coverage of the whole aircraft.

2. MAINTENANCE RECORDER SYSTEM

To fulfill the requirements, different solutions have been considered and the final choice has led to a system (see fig.1) whose basic philosophy is to have an on board recording facility on each individual aircraft of all relevant parameters and an on ground equipment performing two levels of analysis:

- a quick look check to detect the exceedances of design limits for airframe and engine, and the possible malfunction in the aircraft systems, to be performed during the turn around time
- a detailed calculation of the fatigue loading experienced during the last flight or small number of flights and the residual life evaluation to be performed, due to the complexity of the calculating routines, on a remote computer facility located in the air base.

In detail, and in particular for the airframe, the reasons for the choice and the identified monitoring procedures are presented.

2.1 AIRFRAME

2.1.1 Method selection

To find the most suitable solution to satisfy the requirements from the airframe side, the possible applicable methods have been compared:

- "g" counters installation
- strain gauges installation
- indirect method.

A brief analysis of each possible solution shows the reasons for the final choice:

a) the normal practice to check the structural life of the combat aircrafts, has been, until now, to rely on g-counters installed on each airplane; these g-counters, record just the number of times when established levels of normal acceleration are exceeded and are practically of no use for an aircraft as TORNADO with a variable wing sweep, large range of masses, extended flight envelope, big variety of external stores and aerodynamic configurations, wide range of different missions, because, the same level of normal load factor causes very different loads on the same component being at the same time function of mass, sweep, mach number, altitude, stores configuration, flap-slat position etc.
As an indication, two self explaining examples can be taken:

- considering a steady manoeuvre giving 5g measured at the aircraft centre of gravity for the same: mass, wing fuel status and altitude, the bending moment at a wing station close to the pivot point (wing root) shows, depending on wing position and Mach number, differences from a minimum of 11% up to a maximum of 33%, these data being derived from in flight measurements.
- considering a steady manoeuvre giving the 70% of the design vertical bending moment at the front fuselage transport joint, this value is related to load factors which differ for more than 30% just according to wing sweeps and Mach number changes.

The differences are very remarkable and due to the fact that the fatigue endurance of a structure is extremely sensitive to the average loads level, it is therefore clear that the g-counter indication cannot be related with acceptable accuracy to the loads acting on the main structure.

b) the ideal case, would be to record directly the stresses experienced in all the major components, through a direct reading of strain gauges; on this bases, in fact, a direct calculation of damage and residual life is possible with high reliability. But, against this solution a series of problems difficult to resolve in an operational environment raises:

- great number of strain gauges requirement with consequent installation problems

- necessity to perform strain gauges calibration on each aircraft and a series of check calibration during the operational life of the aircraft

- necessity to perform a pre-flight check to establish the datum line and to check the correct working of each strain gauge

- necessity of high accuracy in the strain gauges positioning: in fact their extreme sensitivity can cause, on different aircrafts, subtantially different answers under the same load and in the same nominal position

- the relative low life of the strain gauges, compared to the operational life of the aircraft, implies a heavy maintenance activity

- a very high sampling rate is required for the strain gauges readings than affecting the recording capabilities in term of number of parameters.

The strain gauges use, is a typical experimental technique for prototype and development flying but, for the above reasons they cannot be practically used in an operational fleet.

c) the indirect method is based on the recording on each aircraft of the fleet of a certain number of significant flight parameters and on the subsequent elaboration of the data in order to obtain the loads in the most significant parts of airframe main components then allowing for the proper fatigue considerations.
Such a method can fully satisfy the requirements because:

- allows the quick look check based on the recorded parameters

- allows for the fatigue calculation with the necessary accuracy and reliability

- avoids the use of a model investigating statistically the occurrence and the distribution of damage for the fleet during the operative life, being this method not sufficiently accurate and sometime very misleading in the results.

2.1.2 Parameters recording

The parameters proved to be necessary for the required analysis on the airframe and therefore recorded in flight, are subdivided into four groups:

- Aircraft configuration parameters: all those allowing to identify the mass and the centre of gravity position as fuel weight, stores type and location, wing sweep angle or the aerodynamic configuration as wing sweep angle, high lift devices position, etc.

- Air data and aircraft attitude parameters: Mach number, altitude, speed, incidence, sideslip, etc.

- Control surfaces deflections: spoilers, tailerons, rudder, airbrakes, etc.

- Aircraft response parameters as linear and angular accelerations and angular velocities.

A sampling rate of 8S/sec, as a minimum, is required for all the aircraft response parameters, air data and aircraft attitude, primary control surfaces deflection; this because such a sampling is suitable for the representation of the rigid body motion of the aircraft; for other parameters 1S/sec or 0.5S/sec is enough.

The data compression procedure is adopted to increase sensibly the recording time capability. Fig.2 shows in detail the parameters required for the structure monitoring.

2.1.3 First analysis

The purpose of the first analysis is to supply clear and complete informations about the aircraft use during last flight, to enable a quick evaluation of the experienced loading levels and their comparison with the allovable structural limitations, to detect the possible exceedances of the above limitations and to finalize the decision to go for next flight with the required safety.

The analysis is performed during the turn around time (average duration of 15 minutes), near the aircraft, with a proper equipment identified as Mobile Quick Look Facility (it is hand carried and reasonably light weight), able to read the cassette from the recording unit and to

process the data according to the criteria established for check. It is intended to base the GO/NO GO decision on the check of: vertical loed factor, roll rate, Mach number, calibrated airspeed, sinking speed.

A limited number of parameters (fig.3) is selected from the complete structural list to identify the characteristics ruling the limitations, these being function of the wing sweep, the high lift devices position, the mass of the aircraft, the fuel status in the wing, the external stores configuration and, of course the Mach number and the speed. A credibility check is performed on the selected parameters to detect the possible spikes and/or uncorrect values; each parameter is compared with its possible range and max rate of change to ascertain its validity. Afterward, the procedure for the analysis is slightly different depending on the aircraft configuration: clean or with external stores.

2.1.3.1 Clean aircraft

a1) Vertical load factor
The vertical load factor is recorded in flight and is compared with the allowable only when it is higher than a minimum imposed value, for speeding up the check.
To identify the allowable load factor it is necessary to know:

- wing sweep angle : directly recorded (four positions are considered: fully open, two intermediate, fully swept)

- flap and slat position : directly recorded (four positions are considered: cruise, manoeuvre, intermediate, high lift)

- aircraft mass : derived by calculation from recorded data (fuel content directly recorded and the mass of the empty aircraft, input data, are used)

- fuel status in the wing : derived by calculation from recorded data (fuel content directly recorded and the fuel transfer sequence, input data, are used)

- roll rate : directly recorded.

Each combination of wing sweep angle, flap and slat position and fuel status in the wing, identifies the allowable symmetric load factor as function of the aircraft mass; the structural capabilities of main components are imposing this allowable which will then protect the whole aircraft. Important role is played by the roll rate: in fact a reduction of the allowable symmetric load factor is applied according to the value assumed by this parameter during the in flight manoeuvering; 90% or 80% of the maximum are related to intermediate or high values of roll rate. To cover the gap between the different load factor functions when the wing sweep angle or the flap and slat position are changing and when fuel is being transferred from wing to the fuselage, the assumption to compare the measured load factors with the most critical case between the starting or the final position has been made to consider the change of loading conditions due to the varying aerodynamic and/or inertia characteristics.

b1) Roll rate
The maximum value of the roll rate is a design case for the wet wing outboard structure, then the check is required for each asymmetric/rolling manoeuvre performed with fuel in the wing. The roll rate is recorded in flight and is compared with the allowable which, being function of the wing sweep only, can be identified with:

- wing sweep angle: directly recorded

the fully open and the first intermediate positions only can be critical and are then checked.

c1) Mach number
The Mach number is recorded in flight and is compared with the allowable, function of wing sweep and flap/slat position; the parameters required are both directly recorded:

- wing sweep angle

- flap and slat position

Each relevant flight envelope Mach/altitude can be easily checked.

d1) Calibrated airspeed CAS
The importance of the Calibrated airspeed for the design of the structures requires a check for this parameter; it is dependent on the wing sweep and flap/slat position, then the comparing procedure is similar to that at point c1).

e1) Sinking speed
The sinking speed is the most significant parameter to have the indication of the loading level on the undercarriage; for this reason it is derived by calculation using:

- vertical velocity : directly recorded
- ground velocity : derived by calculation from North and East velocities and heading angle (directly recorded)
- derotation rate : directly recorded
- runway slope : input data
- aircraft mass : derived by calculation (see point a1))
- undercarriage on ground : directly recorded
- thrust reverser on : directly recorded.

To have the final allowable, corrections are made proportional to the bank angle to consider the asymmetric landings. Very important, for the nose undercarriage are the derotation speed and the thrust reverser effects. Fig.4 shows the check procedure for the clean aircraft.

2.1.3.2 Aircraft with stores

The presence of the external underwing or underfuselage stores implies some additional limitations to those of the clean aircraft; they are peculiar of the store type and status (e.g. fuel tanks) and sometime of the store position. A very big effort has been spent to be able to record the initial stores configuration and to have, during the flight, the continous updating of such configuration.

a2) Vertical load factor
The same procedure as for clean aircraft except that:

- the aircraft weight is derived considering the pylons and stores weights in addition to the empty mass and the fuel content
- the allowable load factor is dependent on the single stores and their combination and is subject to changes during flight together with the configuration evolution. The limitations are superimposed to the clean aircraft ones and the minimum between the two is the final allowable
- the fuel content is now the sum of internal and external tanks; the fuel transfer sequence gives the informations about the location
- when a store is released/jettisoned the flight load factor at the release/jettison time is checked with the relevant allowable.

b2) Roll rate
Same procedure as for clean aircraft.

c2) Mach number
Same procedure as for clean aircraft but using the Mach limits relevant to the stores configuration. Additional check for Mach limits in release/jettison cases, is provided.

d2) Calibrated airspeed CAS
Same procedure as for clean aircraft but using the CAS limits relevant to the stores configuration. Additional check for CAS limits in release/jettison cases, is provided.

e2) Sinking speed
Same procedure as for clean aircraft with different procedure to define the aircraft mass (see point a2)).

Fig.5 shows the check procedure for the aircraft with stores. When the analysis of the flight is completed the results shown by the MQLF can be:

- GO indication

- NO GO indication

- WARNING indication

the GO indication is presented when no limit has been exceeded during the complete flight, and the permission for the next one is automatically released.

The NO GO indication is presented when one or more limits have been exceeded during the flight, at the same time or at different times; in this case the permission for the next flight is subject to the analysis of the exceedence (all the relevant parameters necessary for the analysis are shown by the MQLF for the time of the exceedance) and, if necessary, different levels of check on the structure. The Warning indication is presented when:

- there is difference between operational and design limits and the measured parameter belongs to this gap

- the measured parameter belongs to a tolerance band on the design values applied to cover the unaccuracy in the measurements of the parameter itself and of the parameters used to establish the allowable.

In the case of the warning the release to next flight is subject to the judgement of the technical authority of the air base.

2.1.4 Detailed analysis

The purpose of the detailed analysis is to evaluate the damage experienced by the airframe during each flight and to establish the remaining life. The requirement is that the time of the analysis must be one half of the flying time; the analysis is performed by a computer facility named Automatic Ground Station whose main task is the fatigue consumption calculation; additional utilities allow the AGS a series of data elaboration and presentation for additional investigations and/or purposes.

The complete set of parameters recorded in flight, is used for the detailed analysis; after the credibility check, the flight is fully analyzed selecting the time slices that can be significant for the fatigue calculation (mainly those with high rate of change of some reference parameters). The loads are calculated in these time slices and then selected again to give the significant points of each manoeuvre thus allowing the change of the loads time histories into loads duty cycles to be added to the already cumulated ones.

The obtained spectra are directly compared with those tested in the fatigue test presently performed on the complete aircraft or furtherly transformed into stress cycles, where the test results are not available, and then compared with the design spectra. The damage calculation for each fatigue critical component and the residual life determination, is the final result of the analysis.

The monitored components (presently around 10 critical items are considered), are located on: wing, tailerons, fin and fuselage, thus covering the whole airframe. If the results of the above mentioned fatigue tests will show some unexpected critical item, it will be easily included in the monitored group replacing a less critical one.

The loads are derived from the flight loads measurements obtained from the Flight Loads Survey program extensively performed on fully instrumented TORNADO prototypes and covering all the main structural components: data are available for wing, fin, taileron, front and rear fuselage, external underwing and underfuselage stores, thus allowing the definition of the load levels with the necessary accuracy. Due to the very high influence of the load levels accuracy in the fatigue life determination, particular attention has been paid for this flight measured data bank.

If the case of loss of recorded data will occur, the flight will be covered by data theoretically derived with design spectra. With this analysis, the confirmation of the structural

integrity is obtained and the fatigue endurance of the airframe is defined, thus satisfying the requirement of a long term safety and, at the same time, giving important informations about the usage of each aircraft to optimize its operational life. Fig.6 shows the check procedure for the airframe.

2.2 ENGINE

The engine monitoring system is mainly aimed to evaluate the low cycle fatigue (L.C.F.) life consumption of the group A parts, that is, components which have a finite life and whose failure could result in a catastrofic damage to the aircraft. The basic parameters needed for L.C.F. usage monitoring are grouped as follows:

- Aircraft parameters : pressure altitude, calibrated airspeed, total air temperature directly recorded
- Engine (left and right) parameters : HP, LP, IP rotor speed, Turbine Blade temperature directly recorded
- Additional parameters : HP, LP, IP Torques and Stator Outlet Temperature derived from the basic ones using performance curves.

For the L.C.F. purposes, a sampling rate of two per second is considered adeguate. After the credibility check, the flight recorded data, will be processed by a computer programme on the AGS which will performe, for all critical areas of group A parts, the following tasks:

a) calculation of stresses (centrifugal, thermal, torsional etc) and temperature combination occured during the whole mission.

b) reduction of the flight profile by successively removing each complete minor cycle and translation of sub-cycles into equivalent zero-max cycles.

c) relative damage factor evaluation by using the material S-N curves.

In addition, the continous recording of engine parameters, will allow to performe on the MQLF a check to provide GO/NO GO informations. The quick look analysis will allow for:

a) check of engine starting

b) check of setting limit excedences

c) time-temperature exceedences

d) vibrations, oil pressures warning.

The list of parameters required for L.C.F. and the quick look check is shown in fig.7.

2.3 AIRCRAFT SYSTEMS

Selected parameters of aircraft systems are recorded in flight, and, integrated with those displayed on the Central Maintenance Panel (CMP) are interpreted on ground by the MQLF for a rapid GO/NO GO indication. The recorded data will be subsequently analyzed by the AGS with the aim to generate an up dated situation of LRU's and systems failures for each aircraft to be used to plan repairs or replacementes and, when possible, to feed back to aircraft component manufacturers precise informations on product quality. The recorded parameters are shown in fig.8. In addition 80 parameters (ON-OFF LIGHTS) are taken from CMP.

3. CONCLUSIONS

The proposed System will record during the flight some 75 parameters relevant to airframe, engine and systems monitoring. Considering the sampling rate required for each parameter and varying from 0.5 to 8S/sec, a total number close to the Data Acquisition Unit capability of 256 words/sec will be recorded; in addition, 80 parameters giving information from the CMP will be added. The realization of the system is at its final stage, being now fitted on a series aircraft to undertake the installation and EMC testing; the flight testing will follow in 84 and after the final clearances issue, it will start the in service activity beginning 85 on the Italian TORNADO, representing a surely original and reliable contribution to the control of aircraft operational life for the flight safety aspects.

MAINTENANCE RECORDER SYSTEM

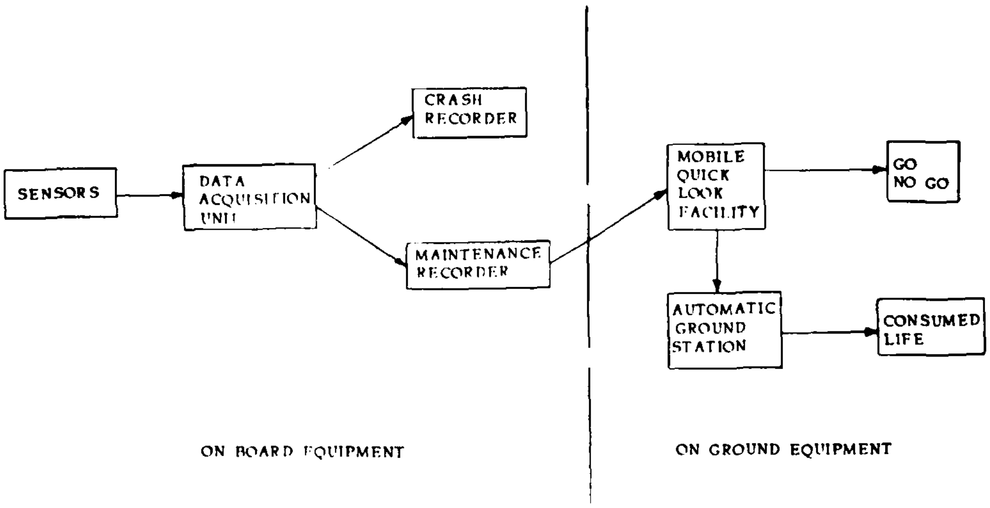

Fig. 1

PARAMETERS
FOR
STRUCTURE MONITORING

Aircraft configuration	Fuel content Wing sweep angle Flap and slat angle External stores: initial configuration (type & position) updating configuration (jettison/release) U/C touch down
Air data and aircraft attitude	Mach number Calibrated air speed CAS Altitude Incidence Sideslip (derived by calculation) Inclination angle Heading angle Bank angle North, East and Vertical velocities
Control surfaces deflections	Spoilers angle Taileron angle Rudder angle Airbrakes position Thrust reverser
Aircraft response	Angular rates p, q, r Linear accelerations n_y, n_z Angular accelerations p, q, r

FIG.2

PARAMETERS
FOR
STRUCTURE FIRST ANALYSIS

Aircraft configuration	Fuel content Wing sweep angle Flap and slat angle External stores: initial configuration (type & position) updating configuration (jettison/release) U/C touch down
Air data and aircraft attitude	Mach number Calibrated air speed CAS Altitude Heading angle Bank angle North, East and Vertical velocities
Aircraft response	Angular rates p, q Linear accelerations n_z

FIG. 3

AIRFRAME FIRST ANALYSIS

CLEAN AIRCRAFT

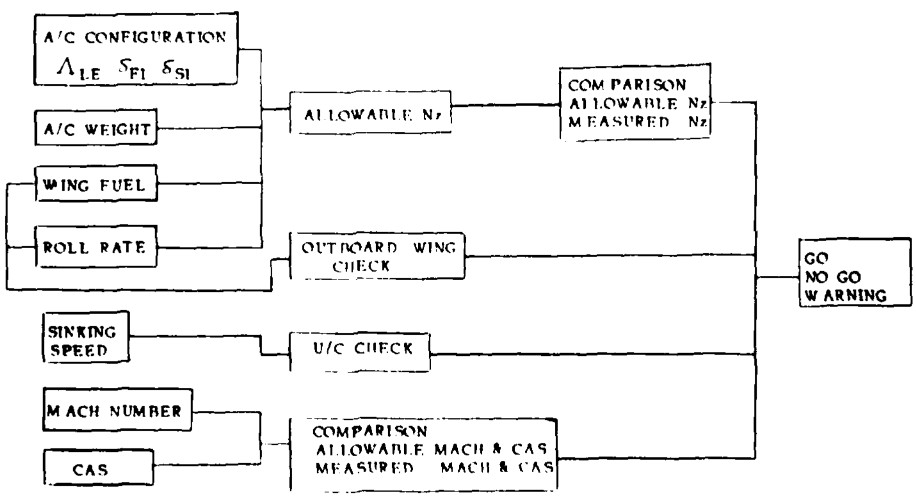

Fig. 4

AIRFRAME FIRST ANALYSIS

AIRCRAFT WITH STORES

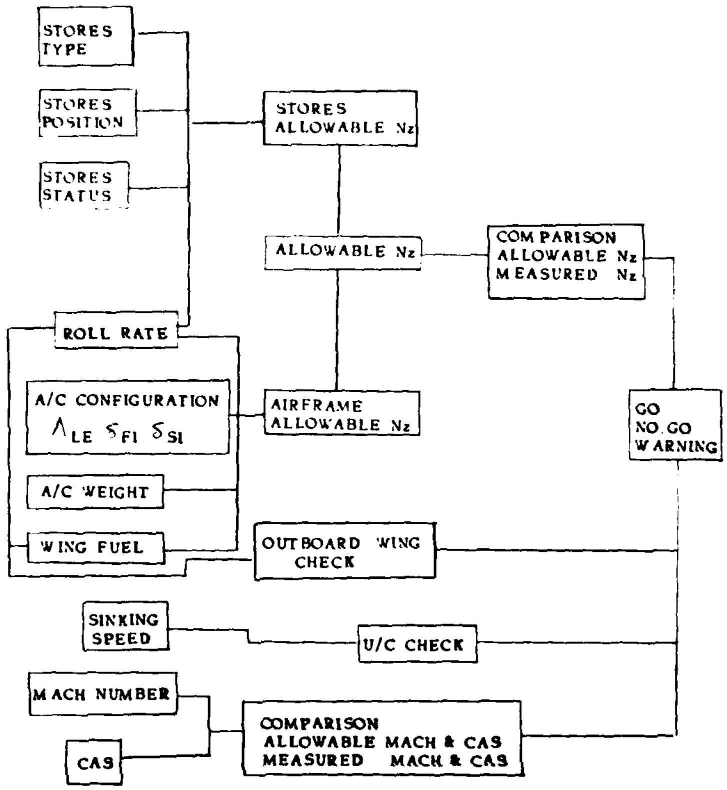

Fig. 5

AIRFRAME DETAILED ANALYSIS

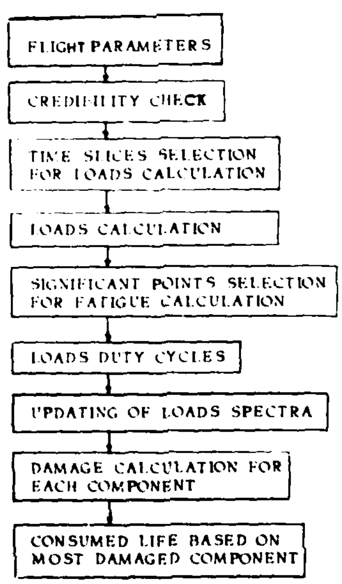

Fig. 6

PARAMETERS
FOR
ENGINE MONITORING

PARAMETER	L.C.F.	Q.L.
PRESSURE ALTITUDE	X	X
CALIBRATED AIRSPEED	X	X
TOTAL AIR TEMPERATURE	X	X
HP SPOOL RPM	X	X
IP SPOOL RPM	X	X
LP SPOOL RPM	X	X
TURBINE BLADE TEMPERATURE	X	X
NOZZLE AREA		X
POWER LEVER POSITION		X
TURBOT WARNING		X
VIBRATION WARNING		X
OIL PRESSURE WARNING		X
BUZZ WARNING		X

FIG.7

PARAMETERS
FOR
A/C SYSTEMS MONITORING

1. OLEO-SWITCH
2. INBOARD SPOILER POSITION, STARBOARD
3. OUTBOARD SPOILER POSITION, PORT
4. RUDDER POSITION
5. TAILERON POSITION, STARBOARD
6. TAILERON POSITION, PORT
7. WING SWEEP ANGLE
8. 115 VAC 400 Hz, 1 Ø
9. 28 VDC
10. FLAP SLAT ACTUATION
11. ICE WARNING
12. FIRE WARNING, PORT
13. FIRE WARNING, STARBOARD
14. ENGINE VIBRATION PORT RED
15. ENGINE VIPRATION STARBOARD RED
16. HYDRAULIC CONTROL LEFT
17. HYDRAULIC CONTROL RIGHT
18. CABIN PRESSURE
19. FUEL LOW WARNING
20. CRASH & MAINTENANCE RECORDER DAU FAULT

FIG.8

ROTORCRAFT ICING TECHNOLOGY - AN UPDATE

by

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AD P002702

SUMMARY

Flight testing of rotorcraft in natural icing conditions is essential to the US Army Airworthiness Qualification requirements and to the US Federal Aviation Administration certification requirements as well as to support research and development efforts. Complete qualification/certification under natural icing conditions only is very expensive in terms of time due to the limited availability of the required test conditions in the natural environment. Consequently, inflight icing facilities that can duplicate the natural environment as well as extreme test conditions rarely found in natural conditions but necessary to meet Airworthiness Qualification/Certification requirements are essential to augment natural icing tests. It is essential that such facilities accurately duplicate the natural environment so that test results can be used for Airworthiness Qualification/ Certification with a high degree of confidence.

1. INTRODUCTION

The US Army has conducted artificial and natural icing tests on rotorcraft since 1973. Flight tests have included research and development efforts as well as airworthiness qualification testing of US Army aircraft for flight into icing conditions. Artificial icing tests have been conducted using the Canadian National Research Council (NRC) icing spray rig in Ottawa, Canada, and the US Army JCH-47C with an installed Helicopter Icing Spray System (HISS). The HISS has proven to be a very valuable test facility, not just for the US Army but for other agencies such as the National Aeronautics and Space Administration (NASA), the Federal Aviation Administration (FAA), US Air Force (USAF), and United States aircraft manufacturers. A continuous development effort has resulted in significant improvements in the HISS, and the US Army icing flight test capability has expanded to improve qualification efforts. The improvements have ultimately led to an artificial spray cloud which has characteristics approaching those in the natural icing environment. Future efforts will ultimately result in a new system which will significantly improve inflight artificial icing test capability.

Many of the US Army's efforts in rotorcraft icing technology coincide with Advisory Group for Aerospace Research and Development (AGARD) efforts to influence technology. Specifically, many of the AGARD recommended efforts contained in AGARD Advisory Report No. 166, Rotorcraft Icing, Status and Prospects, August 1981, have been addressed by the US Army and are reported on in this paper as an update of rotorcraft icing technology. While not all inclusive, the following selected programs provide an update of some flight testing conducted to date relative to the "icing test facilities" and "ice protection systems" discussions contained in AR 166:

a) Flight testing of pneumatic deice boots installed on a UH-1H helicopter in artificial icing conditions behind the Canadian NRC Ottawa Spray Rig.

b) Artificial and natural icing tests of the unprotected CH-47C and UH-60A helicopter rotor systems.

c) Evaluation of ice shapes on a UH-1H rotor system and hover performance degradation caused by rotor icing.

d) Current and Planned improvements to the US Army JCH-47C Helicopter Icing Spray System (HISS).

The US Army currently qualifies aircraft for flight through the moderate icing intensity, which is defined as 1 gm/m^3 liquid water content (LWC). Based on present design criteria of protecting rotorcraft to 1 gm/m^3 there is less than a one percent probability that more severe icing conditions will occur on a typical operational mission in continuous maximum (strat form clouds) icing. The criteria is fully defined in reference 1, and shown in figure 1. While the US Army qualifies to 1 gm/m^3 at the current time, future plans are to increase the HISS capability to produce at least 2 gm/m^3 and, if possible, 3 gm/m^3. This will allow a more thorough evaluation of intermittent maximum icing conditions on the rotorcraft's capabilities as well as provide an improved research and development capability.

2. FLIGHT TESTING OF PNEUMATIC DEICE BOOTS

The NASA initiated a program in 1980 to evaluate the feasibility of pneumatic boot deicing concepts for helicopter rotor systems. The B.F. Goodrich Company (BFG) proposed a concept which used a polyurethane elastomeric material (trade name, Estane) for the boot. Limited wind tunnel icing tests conducted by NASA's Lewis Research Center indicated that it might be feasible to produce a pneumatically deiced rotor blade. NASA's Ames Research Center requested the assistance of the US Army Aviation Research and Development Command (AVRADCOM) in flight testing the concept. Subsequently, AVRADCOM tasked the US Army Engineering Flight Activity (USAAEFA) to conduct feasibility testing of the Pneumatic Boot Deicing System (PBDS) concept. Potential advantages of pneumatically deiced rotor systems include a low power requirement, reduced system complexity, reduced weight, and rapid deicing of the rotor blade.

BFG manufactured a set of pneumatic boots and installed them on a set of U.S Army-owned UH-1H main rotor blades. Bell Helicopter Textron (BHT), under a NASA contract, instrumented one blade, main rotor hub, mast, pitch change link, and the three main rotor control actuator extension tubes for inflight measurement of structural loads. BHT also designed and built an instrumentation and pneumatic slip ring assembly to be installed at USAAEFA prior to testing.

The objective of this program was to determine an operational envelope and conduct feasibility testing of the PBDS concept in both dry air and artificial icing conditions. (Ref 2)

The test aircraft was a JUH-1H helicopter U.S. Army S/N 70-16318. It was modified to incorporate a partial ice protection system, which included pneumatically deiced main rotors, heated glass windshields and a Rosemount ice detector. The aircraft was also instrumented to record structural load, performance, and handling qualities parameters. A schematic of the pneumatic deice system and cross section of the deice boot is presented in figures 2 and 3, respectively. Basically, bleed air from the customer bleed air source on the engine is routed through a regulator valve then up the main rotor mast to a pneumatic slip ring and then to the leading edge pneumatic boots. The boots are deflated by forcing engine bleed air through an ejector pump to apply continuous suction to the rotor blades except during the inflation cycle.

Flight testing to develop a flight envelope included ground runs, hover, climbs, level flight, turns, descents, and autorotational flight. The boots were inflated, deflated, and partially inflated as would be typical of the case if the air source for the ejector pump was turned off, or the engine failed. All structural loads were well within the prescribed limits for the limited envelope developed except for the main rotor pitch link axial load which exceeded endurance limits during turns of 30 degrees of bank with the boots inflated. The flight envelope for the pneumatic system was opened up to 100 knots (185 km/hr) calibrated airspeed (KCAS) and a gross weight of 8600 pounds (3900 kg).

Investigation of handling qualities has shown that there has been no significant difference between the standard UH-1H and the test JUH-1H with the pneumatic boot deflated. Inflation of the pneumatic boot, however, resulted in a mild right yaw, followed by a right roll and pitch down. All rates were easily compensated for by the pilot.

Performance testing revealed substantial performance penalties associated with the pneumatic boot both in the inflated and deflated mode. Nondimensional hover performance at a five-foot (1.5 m) skid height is presented in figure 4. The data compares a standard UH-1 to the test aircraft with the pneumatic system inflated, deflated (normal), and with the system vented (partial inflation) such as would occur, if the ejector pump air source was interrupted.

Level flight performance is shown in figure 5. At 90 knots (167 km/hr) true airspeed (KTAS), the pneumatic boot system requires approximately 150 additional shaft horsepower if the system is deflated and a 250 shp increase if the system is vented.

Icing tests were conducted in Ottawa, Canada at the NRC spray rig shown in figure 6. The test conditions were 0.25 gm/m^3 at temperatures of -6 to -20°C; 0.4 gm/m^3 at -9 and -11°C 0.5 gm/m^3 at -6, -11, and -15°C; and 0.75 gm/m^3 at -6, -10 and -11°C. Initial evaluation of the pneumatic deice boots is encouraging. Except for small amounts of residual ice, activation of the system cleaned the main rotor blades of accreted ice as shown in the photos in figures 7 and 8. The pneumatic system shed ice successfully with the rotors turning at a hover and also without the aid of centrifugal force and vibration when the blades were stationary. The aircraft should be ready for forward flight icing tests during the 1983/1984 icing season.

3. UNPROTECTED ROTOR SYSTEMS

The USAAEFA has conducted two flight test programs to evaluate the abilities of unprotected rotor systems to operate in icing conditions. The CH-47 and UH-60 helicopters were the test articles. In both programs, the aircraft were equipped with electro thermal deice systems on the rotors, but the systems were not activated unless an inflight situation arose which required deice of the rotor system. Both artificial (to gain confidence in procedures and equipment) and natural icing tests were conducted. Test conditions for the UH-60 test are presented in figure 9. The test conditions

were documented by a U-21 fixed wing aircraft which recorded liquid water content, outside air temperature, median volumetric diameter and relative humidity. The U-21 cloud measuring equipment is described in detail later.

Main rotor ice sheds occurred on the UH-60 in both artificial and natural icing environments. The first ice shed during a flight typically occurred from 15 minutes to an hour after entering the icing environment. Random and frequent sheds occurred thereafter. The main rotor ice sheds resulted in light to moderate increases in airframe vibration which normally lasted less than a minute. This increased vibration level was barely apparent to an experienced aircrew fully occupied by their tasks and, on occasion, noticeable only if their attention was directed to it. One main rotor ice shed occurred during artificial icing at 0.5 gm/m^3 LWC and -12.5°C, which resulted in an airframe vibration immediately apparent to the aircrew but did not significantly affect their work load over the length of time the vibration lasted (approximately 10 minutes). The increase in airframe vibrations due to main rotor ice shedding was not a problem.

Engine torque increased with ice accretion and decreased when ice was shed from the rotor system. Accurate measurement of torque increase/decrease was not possible while flying in the artificial icing environment due to constant collective control changes required to fly formation and keep the main rotor positioned in the artificial cloud; therefore, an estimate of the percentage of torque rise in the artificial icing environment could not be made. A fixed collective setting was used while in the natural icing environment permitting a more accurate correlation between torque increases/decreases and rotor blade ice accretion/shedding. The largest torque rise was observed in natural icing conditions at 0.5 gm/m^3 LWC and -4.0°C; however, major engine damage occurred during this flight due to ice ingestion from a rotor ice shed as discussed later. Turbine gas temperature (TGT) remained within the normal continuous limits during power increases at test gross weights and altitudes. However, at heavier gross weights in similar icing conditions, the TGT might rise to the 30-minute limit range or could reach the maximum TGT limit, resulting in loss of rotor speed.

The UH-60A unheated droop stops and flap restrainers were evaluated throughout the icing tests for proper positioning during main rotor shutdown. Droop stops freezing in the fly position allow excessive blade droop during engine shutdown. This occurred on several natural icing flights where the droop stops were not heated. This reduced airframe and blade clearance, and could cause the blades to strike the fuselage or ground personnel especially while shutting down during gusty wind conditions.

The aircraft sustained damage during flights under both artificial and natural icing conditions. During an artificial icing flight at 0.49 gm/m^3 LWC and -6.0°C, a tail rotor blade tip cap was damaged, requiring replacement. During another artificial icing flight at 0.5 gm/m^3 LWC and -12.5°C, the white strobe portion of the upper anticollision light was broken by ice shed from the main rotor. The shattering glass from the anticollision light was thrown into the tail rotor blades causing damage to the leading edge of all four blades; and, the tail rotor gear box fiberglass cowling was hit by ice shed from the main rotor causing a dent and split which required repair of the cowling.

The final test flight was in natural icing conditions of 0.5 gm/m^3 LWC and -4.0°C. A 17 percent torque rise per engine occurred after the aircraft had been in the icing environment 13 minutes. At 29 minutes, a main rotor ice shed occurred and torque decreased to the preimmersion trim power required for cruise (collective fixed). Within 5 minutes after this shed, the torque had increased 14 percent per engine with random main rotor ice sheds occurring every 3 to 5 minutes, resulting in torque decreases of 3 to 4 percent. No significant increase in vibration level was noted during the ice sheds. After 43 minutes in the icing condition, torque was 18 percent above trim preimmersion cruise power. At this time, ice was shed from the main rotor and was ingested into the No. 2 engine, causing a rumble, similar to that of a compressor stall, accompanied by a high pitched squeal and TGT increase of approximately 40°C. The icing condition was exited immediately and the aircraft was landed at an outlying airport. This flight produced the highest torque rise and most costly aircraft damage of any icing condition tested. A borescope inspection revealed major damage to the compressor section of the No. 2 engine requiring replacement.

The UH-60 apparently has some capability to operate in light icing conditions without an ice protection system on the main rotor system. The rotor system accreted ice and self shed. Power required for flight increased resulting in range degradation. Vibration levels increased but were not excessive. The aircraft handling qualities and vibration levels were acceptable throughout the limited artificial and natural icing conditions tested. The UH-60A demonstrated safe operation in light icing conditions up to and including 0.3 gm/m^3 LWC without operating the blade deice system. However, without an ice protection system to control the time and size of ice sheds, the amount of aircraft damage was increased. This translates into a higher operating cost required to fly unprotected rotor systems in a light icing environment. (Ref 3)

4. ICE SHAPES AND PERFORMANCE DEGRADATION

The Applied Technology Laboratory (ATL), Research and Technology Laboratories, US Army Aviation Research and Development Command (AVRADCOM), and NASA Lewis Research Center (NASA) are jointly undertaking a program to predict the performance penalties

associated with helicopter operations in icing conditions. The initial phases of the ATL/NASA effort requires gathering hover performance data with both clean and iced rotor blades, and documenting the topography of the ice accretion on the rotor blades. AVRADCOM tasked the USAAEFA to conduct the hovering flight tests on a UH-1H at the NRC Icing Spray Rig.

The objective of this test was to gather comparative hover performance and blade surface topography data for the UH-1H helicopter with both clean and iced rotor blades. Documentation of the ice accretion shapes is to be accomplished by NASA.

The test conditions were at temperatures ranging from-3°C to -21.5°C for a LWC of 0.4 gm/m^3, -8.7 and -7.5 at 0.25 gm/m^3, and -19°C for 0.7 gm/m^3. The test technique was to obtain hover data using free flight method prior to entering the cloud. The free flight technique required variation of gross weight and rotor speed. Data was obtained with the aircraft pointed into the wind, at a skid height of 50 ft (15.2 m).

The test aircraft was then hovered with the main rotor immersed in the cloud. Initially, the advice of experienced spray rig operators was the primary method used to determine the amount of ice accreted on the main rotor prior to spontaneous shedding. As experience was gained test team members were able to identify the optimum immersion time.

After the rotor was iced the aircraft exited the artificial cloud and free hover performance data repeated. The aircraft was moved to the vicinity of the ice documentation station which was within 300 ft (91.4 m) of the spray rig and shut down. The rotor was stopped using a rotor brake applied so as to smoothly stop the rotor. Just prior to the rotor stopping, the rotor brake was released so that backlash of the rotor drive train would not cause ice to shed from the blades.

The Aeronautical and Astronautical Research Laboratory of the Ohio State University (AARL-OSU) was responsible for all ice shape documentation. Main rotor ice accretion was documented using the following techniques:

a) Molds of the entire rotor ice accretion.

b) Steroscopic photographs of the entire rotor ice accretion.

c) Tracings or photographs with the grid background of selected ice accretion cuts. (Ref 4)

The documentation station consisted of a rented airline food service truck shown in the photo in figure 10. The cargo area of the truck was modified with heaters, lights, and electrical power outlets. The ability of the cargo section to be adjusted in height allowed for one rotor blade to be almost completely inclosed in the cargo section by backing the truck up to the side of the UH-1H. This procedure allowed workers to get close to the iced rotor blade, moderated the working environment, and protected personnel from flying ice if another spray rig test was being conducted while ice documentation was in progress.

The primary documentation technique was to mold the ice shape on the leading edge of the rotor shown in the photo in figure 11. This allowed the most straight forward method of reproducing the shapes for wind tunnel tests. The second most desirable method was stero photography. Two cameras were used in conjunction with a grid superimposed on the ice contour. The final method was to cut the ice and then trace a profile, obtain an impression, or photograph an inserted grid.

The flying and documentation phase of this program was conducted January 1983 through March 1983 and consisted of approximately 10 hours of artificial cloud time. The wind tunnel testing, computer coding, and analysis is still in progress.

5. CURRENT STATUS OF THE HELICOPTER ICING SPRAY SYSTEM (HISS)

The US Army HISS is installed in a JCH-47C. It is designed to produce an artificial icing cloud for inflight evaluation of aircraft anti-ice and de-ice capabilities. The HISS consists of an 1800 gallon (6.81 Kg) water tank, plumbing, pump, and spray boom. A photo of the HISS is shown in figure 12. A schematic of the HISS is presented in figure 13 (Ref 5). The water is pumped to manifolds on the boom, which distributes water to nozzles where it is atomized by high pressure, bleed air. The JCH-47C is equipped with a Rosemount sensitive outside air temperature system and a Cambridge dew point hygrometer. A radar altimeter with rear facing antennas is used to position the test aircraft at the proper standoff distance of 150 ft (45.7 m). The HISS has been operational since 1974. It has continually been in a state of modification and improvement. Starting in 1979 major modifications were initiated to improve the quality of the artificial cloud, particularly with respect to the size of the water droplets.

For 1979/1980 icing season, the original All-American Engineering Company (AAE) nozzles were replaced. The median volumetric diameter (MVD) of the droplets generated by AAE nozzles ranged from 100 to 300 micron. The replacement nozzles were type 125-H "Sonicore" designed by Sonic Development Corporation. These nozzles produced a cloud with a MVD more nearly approximately the natural environment, Figure 14 (Ref 6) depicts a comparison of MVD between the artificial and natural clouds. Cloud data was obtained

by mounting particle measuring spectrometers on a UH-1H and immersing them in the artificial cloud. All measured cloud parameters were derived from drop number count, diameter classification, and size of the air volume sampled. There were three probes installed on the UH-1H; an axially scattering probe, a cloud particle spectrometer, and a precipitation particle spectrometer. The data was analyzed by Meterorology Research, Inc. and is presented in detail in reference 6.

Two HISS configurations were evaluated using Sonicore nozzles. Initially 160 nozzles were installed, filling all available center section locations and the inboard three quarters of each outrigger. After three flights in this configuration, the nozzles were removed from the outriggers, leaving 97 nozzles installed in the center sections only. The outriggers were isolated from the boom air and water supply by metal plates bolted between the boom flanges at the outrigger junctions. The boom outriggers were left installed because of boom dynamics considerations. The remainder of the flights used this arrangement. With 160 nozzles, the boom air pressure of 10 psig (0.703 kg/m^2) was less than the minimum needed for satisfactory water atomization. Reducing the number of nozzles to 97 increased air pressure to a nominal 20 psig (1.406 kg/m^2) which was acceptable. The pressure difference is attributed to the resulting change in total air orifice area.

Several other characteristics of the new system were observed during operation. Flow blockage from residual water freezing in the nozzles was greatly reduced by routing bleed air through both the air and water lines (purge) from takeoff until actual start of water flow. This proved effective to temperatures as low as -20°C, provided all residual water had first been eliminated from the boom.

For the 1981/1982 icing season, a solar T-62T-40C2 gas turbine auxiliary power unit (APU) was added to increase air pressure available to the nozzles for improved atomization. Previously the two main engines on the CH-47 were tapped at the engine anti-ice port for compressor bleed air. For purposes of safety and noise reduction, the unit was inclosed in a stainless steel box with fiberglass sound proofing. Bleed air from the APU was ducted to a flow mixer which combined aircraft engine bleed air with APU bleed air. The combined APU and engine bleed air enter the boom through flexible tubing leading to the boom air intake pipes on either side of the cabin. Electrically-operated valves actuated from a single control panel were installed in the system to control both bleed air and water flow rates.

Two HISS nozzle configurations were evaluated after this installation. Initally 165 Sonicore nozzles were intalled to fill all usable locations. Because of air pressure and water endurance considerations, the nozzles were removed from the outriggers leaving 97 nozzles installed in the center nozzles only. The outriggers were isolated from the boom air and water supply in the same manner as before. All artificial icing tests were accomplished in this configuration. With 165 nozzles, the boom air pressure was a nominal 20 psig (1.406 Kg/m^2). This approximates the boom pressure reported in previous years with only the center boom sections operational. Reducing the number of nozzles to 97 increases boom air pressures to between 23 (1.62 Kg/m^2) and 36 psig (2.53 Kg/m^2). This increased pressure permitted slightly better droplet atomization and produced acceptable cloud characteristics at higher flow rates under high relative humidity conditions.

A major problem encountered during these icing tests was the formation of ice masses ("popsicles") on the boom. These ice masses would randomly fall from the boom which posed a hazard to people and equipment on the ground. Leaking water may also increase the MVD of the artificial cloud. Early nozzle configurations (prior to 1980) suffered from this problem, but it had nearly been eliminated during the past two icing seasons. Leakage now occurred where the plastic hoses attach to the brass fittings on the nozzles and the water manifolds. The leaks were probably caused by increased bleed air pressure and temperature from the new APU during the purge cycle. No suitable solution to this problem was found during the icing season and the spray cloud was degraded because of it.

Another recurring problem with the HISS was a difference in spray output between the upper and lower boom sections. Downward flow routing of water from common sources and a difference in static head pressure due to 5 foot (1.5 m) vertical separation resulted in a visible more dense spray emanating from the lower boom. This condition was aggravated by installation of the bleed air APU. Increased bleed air pressure and temperature caused breakdown of hoses and seals within the boom which allowed air to leak into the water system. At low and moderate flow rates, the spray from the upper boom would be intermittent.

Documentation of the cloud properties for 1981/1982 was conducted by a forward scattering spectrometric probe (FSSP) and a optical array probe (OAP) mounted on a U-21 fixed wing aircraft. The probes were immersed in the artificial cloud of 150 ft (45.7 m) behind the spray boom. Vertical sweep measurements indicated higher values of median volumetric diameter (MVD) in the bottom 1/4 of the cloud. This increase was caused by an absence of small drops and not an increase in the number of large drops. The reasons for this shift are probably related to gravitational sorting, bouyancy effects and differences in spray flow rate and atomization between the upper and lower booms.

Figure 15 (Ref 7) shows spray cloud MVD as a function of water flow rate. The MVD data at the lower flow rates fall between 25 and 40 microns. MVD increased as

water flow rate increased. With the increased air supply available, MVD was not expected to increase with flow rate. This MVD increase could be caused by the numerous air and water leaks experienced since the installation of the APU.

Liquid water content (LWC) was obtained from the FSSP and OAP systems. It was backed up by calculated LWC based on the true airspeed, cloud size, and water flow rate. The spray cloud is not homogeneous. The thickest portion of the cloud is just below cloud centerline as shown in figure 16 (Ref 7). The LWC of the cloud is measured in the thickest region. Measured values of LWC as a function of water flow rate are shown in figure 17 (Ref 7). These values are averages of 10 second sample times taken over a one to two-minute immersion. Scatter in the data may be explained by 1 gal/min resolution of the flow meter and measurement techniques (i.e., ability of the pilot to maintain precise position). Test day relative humidity effects the LWC and MVD. In particular, low relative humidity decreases the LWC for a given flow rate and increases MVD by evaporating the smaller droplets.

For the 1982/1983 icing season several reliability and maintainability improvements as well as corrections to the above mentioned problems were accomplished. The water jettison system was redesigned, insulation was added to improve crew comfort, the water and air lines were rerouted to reduce freeze up, and a more reliable water flow measurement system was incorporated. A new manifold system was designed for water distribution in an attempt to reduce leaks and "popsicle" growth. The new manifold consists of stainless steel tubing with solid fittings between the nozzles and the boom as shown in figure 18 (Ref 8). This should cure the external water leak problem of the spray boom. To equalize the flow in the top and bottom spray booms a flow divider was installed. Water to the lower boom was routed down the left boom support and water to the upper boom was routed down the right boom support.

The current US Army artificial icing capabilities using the HISS are:

a) test airspeeds 80 to 120 KTAS (148 to 222 km/hr)

b) liquid water content 0.25 to 1.0 gm/m^3

c) temperatures zero to -20°C

d) MVD 25 to 65 microns

e) Cloud size 8 by 36 ft (2.4 by 11 m)

The current system capabilities are limited by a variety of factors. Maximum test airspeed is limited by engine power available. The minimum airspeed is limited by rotor downwash impingement on the artificial cloud. Flow rates of less than 0.25 gm/m^3 do not fully charge the spray boom with water, and flow from individual nozzles varies. The nozzles are also more likely to freeze at low flow rates. The 1 gm/m^3 limit comes from the fact the Army has no requirements for higher flow rates because the Army only qualifies aircraft to fly into forecast conditions of moderate intensity or less. Additionally, MVD increases significantly at higher LWCs degrading cloud quality. Low humidity conditions cause small drops to evaporate rapidly increasing the MVD of the cloud droplets reaching the test aircraft. At temperatures below -20°C the water system becomes more susceptible to freezing, and crew comfort in the open aft area of the JCH-47C becomes a problem. The current cloud size is approximately 8 by 36 ft (2.4 by 11 m). This normally dictates that only a portion of the test aircraft can be immersed in the cloud at any one time such as rotor system, engine inlets, or fuselage. Gravitational sorting and other factors produce a nonuniform cloud profile. Endurance of the HISS is dependent on both fuel and water expenditures and becomes a factor if long immersion times at high LWC are required. There is approximately 54 minutes of cloud time available at flow rate for one gram per cubic meter The HISS limitation and problem areas have been adequately documented and where possible, corrected. However, many areas still remain which can only be corrected by a complete HISS redesign through future improvements which are now being addressed.

6. FUTURE HELICOPTER ICING SPRAY SYSTEM IMPROVEMENTS

Since 1978, the US Army and the FAA have cooperated to improve the capability of the HISS by executing Interagency Agreements. Through these cooperate efforts significant improvements have been incorporated on the HISS as previously discussed. While the common goal of an improved HISS exists, the current FAA certification requirements and the US Army Airworthiness Qualification requiremennts for flight into icing conditions differ mainly in the severity of LWC (currently the FAA is 3 gm/m^3 which may be revised downward for altitudes below 10,000 ft and the US Army is 1 gm/m^3) for the helicopter. Other parameters, which will be identified later, are in general agreement. The US Army does recognize a need to flight test engine/engine inlets and cowls to 3 gm/m^3 LWC, consequently the requirement to have a HISS capability to produce 3 gm/m^3 is necessary in this case. Additionally, it would allow testing to greater LWC levels should the need arise for additional research and development work and for qualification requirements greater than 1 gm/m^3. Based on known requirements, the US Army negotiated a contract with the Boeing Vertol company in 1982 to conduct a design analysis to determine the design approach for an improved HISS in order to achieve the following spray cloud design goals:

a) Liquid Water Content (LWC)	0 to 3 gm/m^3
b) Droplet Size (medium volumetric)	20 to 50 microns
c) Test Temperature Range	0°C to -30°C
d) Altitude Range (Density Altitude)	0-12,000 ft (3,658 m)
e) Airspeed Range	60 (97 Km/hr) to 150 KTAS (241 Km/hr TAS)
f) Cloud Depth	25 ft (8 m)
g) Cloud Width	75 ft (23 m)
h) Cloud Spray Endurance at 1.0 gm/m^3	1 hour (at 150 KTAS)
i) Aircraft Endurance for conditions above	2 hours

A review of the initial design analysis included air vehicle candidate assessments for the HISS. The candidate assessments are shown in figure 19. Based on availability, capability (including potential growth) and logistical support, the CH-47D was selected as the candidate air vehicle for the new improved HISS. Other candidates were eliminated because of inadequate useful load, airspeed limitation and/or availability/cost/support considerations. In addition to selecting the CH-47D as the air vehicle for carrying the HISS, the type of HISS was defined as a palletized system (a conceptual design is shown in figure 20). The HISS could be installed in the cabin area with the boom assembly actuated through the rear cabin window, for easy removal and reinstallation in another CH-47C/D (a conceptal design is shown in Fig. 21). The current JCH-47D HISS configuration is one of a kind with extensive structural modifications and aircraft hydraulic, pneumatic, and electrical interfacing. When the current JCH-47C is down for any reason so is the HISS and the narrow window for icing qualification test programs in the winter is jeopardized. A palletized HISS which can be removed and installed in another CH-47D and ready for operation within two days eliminates the potential for a lost icing qualification winter season. Unfortunately, achieving all of of the design goals may not be technically feasible with the CH-47D, and prioritizing design goals may be required.

Two major problems exist on the current JCH-47C HISS combination which require resolution before proceeding to a final design. One is the nozzle design (the water droplet MVD is dependent on water/flow rate) and the second is the distortion of the cloud spray pattern due to vortex wake rollup as shown in figure 22. The current nozzle design is not capable of varying water droplet size independent of LWC. A new nozzle development effort may have to be undertaken if a nozzle with this capability cannot be found. The vortex wake rollup problem will be investigated during flight tests by trailing smoke from the current HISS boom at different heights in an effort to determine where the vortex wake will not impinge the spray cloud. The vortex wake rollup phenomenon effectively limits the spray cloud's usefulness to approximately 50 ft (15 m). Flight test data is being evaluated to determine the distance the HISS boom must be extended from the JCH-47C to eliminate interference of the vortex wake with the spray cloud.

During the 1980/1981 icing season artificial and natural icing tests were conducted on the US Army's UH-60A helicopter (reference 9). Several shortcomings under icing conditions were revealed and subsequently corrected. One shortcoming of significant concern with respect to the adequacy of the HISS to duplicate the natural environment was the large increase in power required with ice accumulation on the main rotor system. Figure 23 shows a time history of the UH-60A increase in power required during a deice cycle. An increase of approximately 30% in shaft horsepower required between main rotor deice cycles for relatively benign natural icing conditions of approximately 0.2 to 0.4 gm/m^3 and -11°C is depicted. These power variations in level flight represent an adverse impact on range of 10% or greater. This type of increased power requirement on the UH-60A is rather significant and was not exhibited behind the HISS under artificial icing conditions. Artificial icing tests behind the HISS indicated that for similiar conditions power required increases only half as much. Additionally, when flying the UH-1H under natural icing and artificial conditions such discrepancies in the power required for similiar conditions were not evidenced.

It is suspected that the difference in the HISS water droplet MVD in conjuction with the type rotor blade airfoil section being tested results in ice accretion shapes significantly different between those shapes found in the natural and those seen in artificial environments. The difference in ice shapes was observed by flight test personnel during icing tests. In the artificial icing environment, flight crews normally observed a streamlined conformal ice coating on aircraft airfoil shapes under all LWC and temperature conditions. While in the natural environment the ice form shape was normally nonconforml and exhibited typical horn type protuberences as shown in figure 24. The rotor blade section of the UH-60A is an SC 1095 while the UH-1H is a NACA 0012. The SC 1095 represents an aerodynamically more efficient blade than the NACA 0012 and may therefore be more adversely effected by icing conditions. If this is the case then the validity of artificial icing tests under some conditions behind the HISS may be reduced. The identification of such limitations relative to certification icing tests is essential to the qualification process.

AVRADCOM is currently conducting flight research to identify and document the differences in the ice accretion shapes in the HISS artificial environment and the natural environment. A three phase program will be conducted to accomplish the following:

a) Phase 1. Verification and comparison of the cloud parameter measurements between the USAAEFA U-21A installed system and the US National Center for Atmospheric Research (NCAR) similiarly equipped Beech King Air.

b) Phase 2. U-21A natural icing tests in which the airplane is equipped with the cloud parameter data measurement system and installed airfoil sections.

c) Phase 3. U-21A artificial icing tests in which the airplane is equipped with the cloud parameter data measurement system and installed airfoil sections.

Phase 1 will be conducted with the U-21A and Beech King Air being flown in formation or close proximity under natural icing conditions to verify that the cloud parameter measurements obtained in terms of LWC, water droplet MVD, and distribution characteristics are comparable. This comparison would insure that the values derived could be used in the same data base. Data compared will be obtained from the Forward Scattering Spectrometer Probe (FSSP), an Optical Array Cloud Droplet Spectrometer Probe (OAP), Rosemount OAT sensor, Cambridge hygrometer, Leigh MK-10 ice detector and the Small Intelligence Icing Data Systems (SIIDS). Photos of the equipment are presented in figures 25 and 26.

Phases 2 and 3 will consist initially of identifying, designing and fabricating full scale and partial scale models of the UH-60A SC 1095 and UH-1H NACA 0012 blade section (the span is yet to be determined). These sections will be mounted externally on the U-21A in the free airstream and be adjustable for angle of attack. Heat will be applied to remove ice periodically so that different test conditions can be evaluated expeditiously. Photographic documentation of the ice shape during buildup will be taken continuously. Figure 27 shows a conceptual design of the airfoil sections mounted on the U-21A. The U-21A will be flown behind the HISS in artificial icing conditions and then in natural icing conditions for an extensive matrix of liquid water content and temperatures. Attempts will be made to correlate the ice shape obtained behind the HISS on an airfoil section and in natural ice by scaling using the similitude laws presented in reference 10. Some difficulty in perfect duplication of the ice shapes is expected since the scaling will be based on several variables which must be obtained simultaneously in the natural as well as artificial envrionment. The variables will include LWC, water droplet size, ambient temperature, time in the icing environment and velocity.

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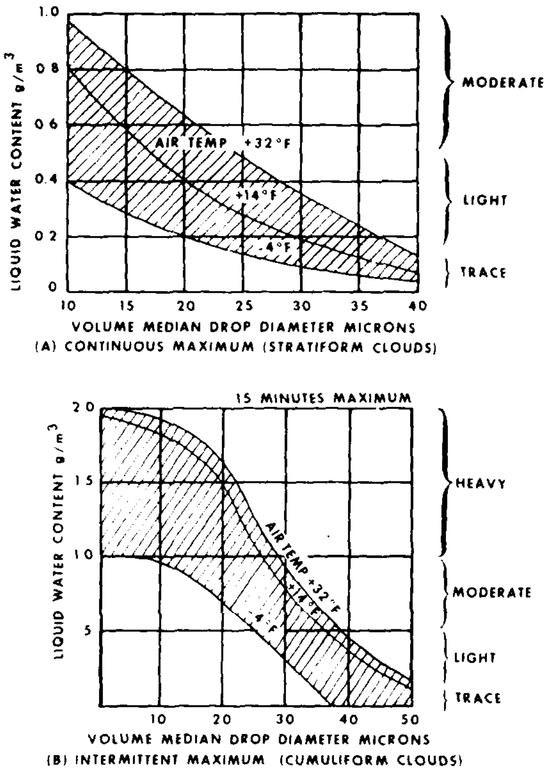

Fig. 1. US Army Accepted Atmospheric Criterion

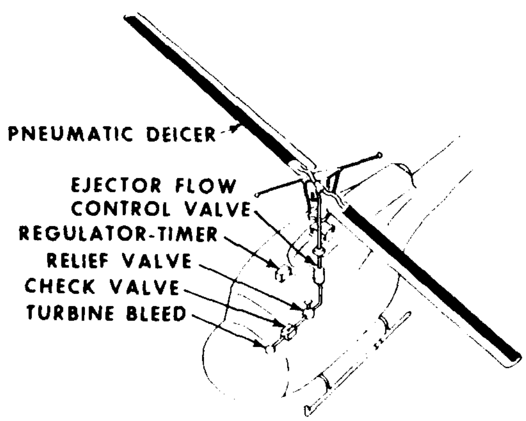

Fig. 2. Pneumatic Deicer System Schematic

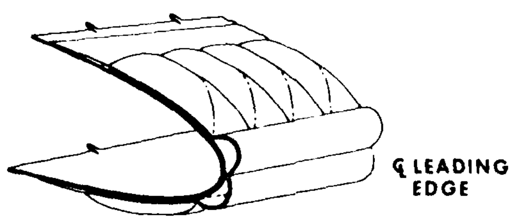

Fig. 3. Pneumatic Boot Cross Section (Inflated)

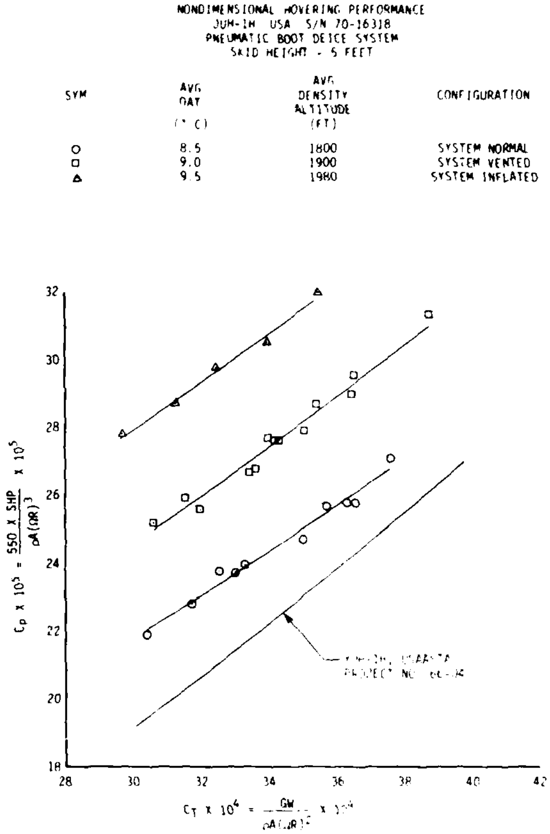

Fig. 4. Nondimensional Hover Performance

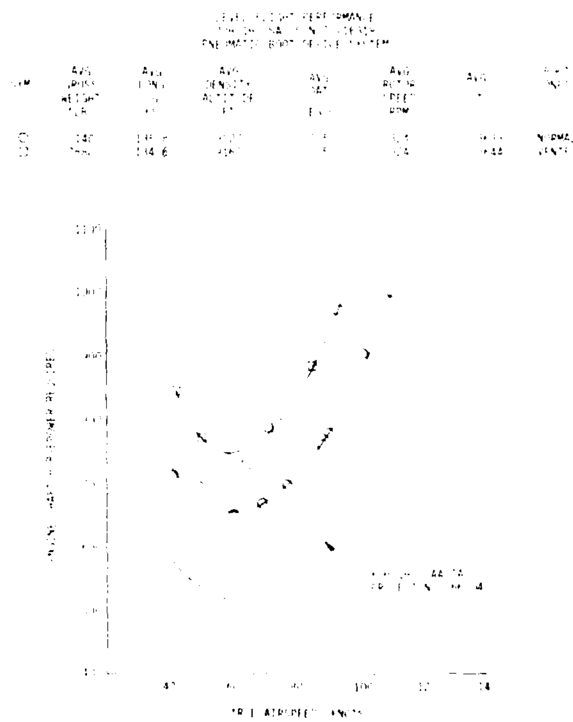

Fig. 5. Level Flight Performance

Fig. 6. UH-1 in Artificial Icing Cloud

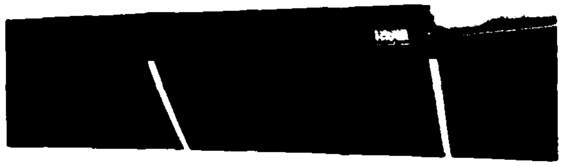

Fig. 7. Ice on Unprotected Portion of Rotor Blade and Ice Removed by Pneumatic Boot Protected Portion Exposure Time 10 Min. LWC = 0.25 gm/m^3 Temp = -10°C

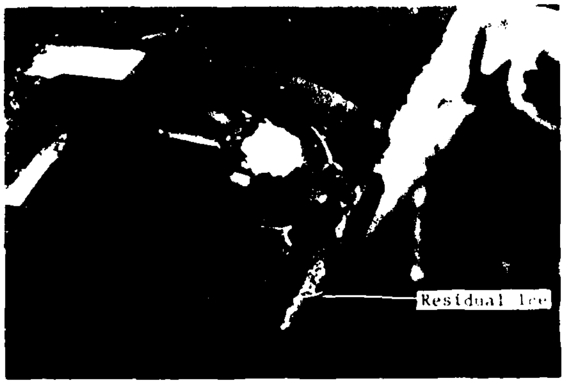

Fig. 8. Residual Ice on Pneumatic Deice Boot

FLIGHT NO.	ICING CONDITION	AVERAGE L.W.C. (GRAMS/CUBIC METER)	MEDIAN VOLUMETRIC DIAMETERS (MICRONS)	AVERAGE RELATIVE HUMIDITY (PERCENT)	AVERAGE O.A.T. (DEG C)	AVERAGE TRUE AIRSPEED (KNOTS)	AVERAGE PRESSURE ALTITUDES (FEET)	TOTAL TIME IN CLOUD (MINUTES)	TOTAL ICE[2] ACCRETED ON VISUAL PROBE (INCHES)	MAXIMUM[3] TORQUE INCREASE PER ENGINE PRIOR TO SHED	RESIDUAL[3] TORQUE INCREASE AFTER SHED	MAXIMUM VIBRATION RATING SCALE
[illegible]	Artificial	0.15	22	47	-7.0	111	5660	60	---	--	--	1
[illegible]	Artificial	0.21	42	10	-17.0	115	7000	45	---	--	--	4
[illegible]	Natural	0.42	20	100	-7.0	115	3120	95	3.25	7	2	4
[illegible]	Artificial	0.50	--[4]	95	-12.0	97	8080	81	---	--	--	6
	Artificial	0.49	82	35	-6.0	115	5040	60	---	--	--	4
[illegible]	Natural	0.30	12	100	-6.0	125	2960	59	2.50	8	4	1
[illegible]	Natural	0.10	9	100	-15.0	117	5280	38	---	--	--	--
		0.20	9	100	-15.0	117	5760	44	2.50	9	5	3
[illegible]	Natural	[illegible]	19	100	-4.0	113	6680	43	3.50	18	0	1

NOTES

Average Gross Weight = 16,260 pounds
Average [illegible] = FS [illegible]
[illegible]tility configuration
[2][illegible]ation of the visual probe precluded it from exposure to the spray cloud during artificial icing.
[3][illegible]rate measurements of torque increases and residual torque increase after a rotor ice shed were not possible during artificial icing due to constant power changes required to maintain position in the HISS cloud.
[4]Data not available

Fig. 9. Specific UH-60 Test Conditions

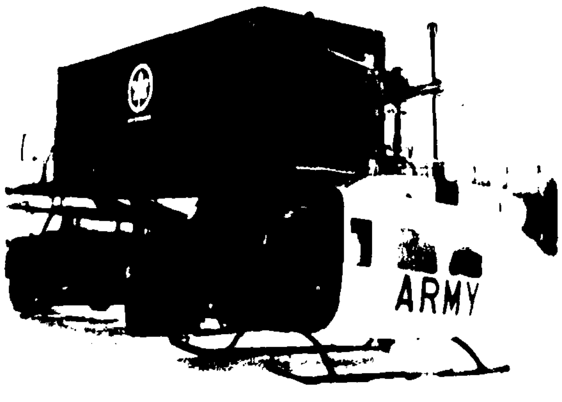

Fig. 10. Documentation Station and Test Aircraft

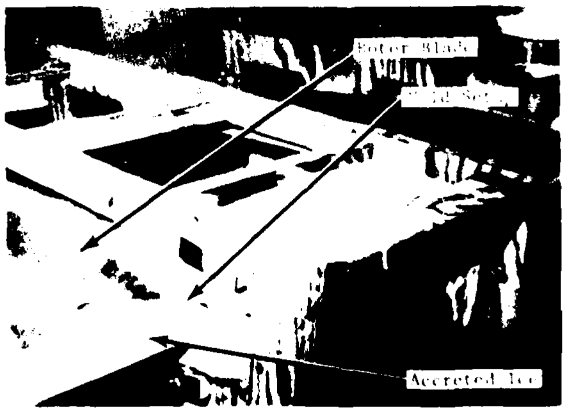

Fig. 11. Inside Documentation Station

Fig. 12. Helicopter Icing Spray System and Test Aircraft in Artificial Ice Cloud

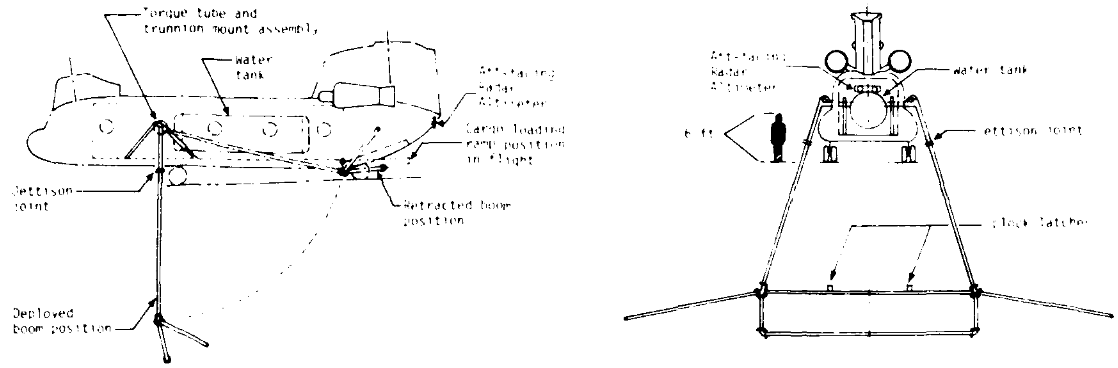

Fig. 13. Helicopter Icing Spray Systems Side and Rear View Schematics

FIGURE 8
COMPARISON OF CLOUD DROP
MEDIAN VOLUMETRIC DIAMETER

NOTES:
1. Present HISS configuration 97 Sonicore nozzles installed on boom center sections only
2. Previous HISS configuration 51 All American Engineering Co. nozzles installed on boom center sections and outriggers
3. Data taken at 90 KTAS
4. HISS cloud depth estimated as 8 ft in present configuration, 10 ft in previous configuration
5. MVD measured with Particle Measuring Spectrometers furnished by MRI
6. Ranges for HISS MVD values are based on combined vertical sweep data from several flights

FIGURE 9
WATER FLOW RATE AND DROPLET SIZE MEDIAN
VOLUMETRIC DIAMETER OF HISS SPRAY CLOUD

SYMBOL	OUTSIDE AIR TEMPERATURE (°C)	PRESSURE ALTITUDE (FT)	RELATIVE HUMIDITY (%)
◇	-10.0 to -13.0	6800 - 9900	10 - 13
◇	- 5.0 to -15.0	3600 - 8900	24 - 29
D	- 8.0 to -19.5	3860 - 9960	55 - 57
□	-15.0 to -17.0	2800 - 9040	60 - 67
◇	-10.0 to -15.0	2500 - 8400	72 - 78
○	- 5.0 to -20.0	6000 - 9240	80 - 85
△	- 5.0 to -15.0	3840 - 9260	90 - 97

NOTES.
1. 97 SONICORE NOZZLES INSTALLED ON BOOM CENTER SECTIONS ONLY
2. DATA SHOWN ARE FOR STABLE IMMERSIONS CENTERED IN SPRAY CLOUD
3. 150 FT STANDOFF DISTANCE, 120 KTAS

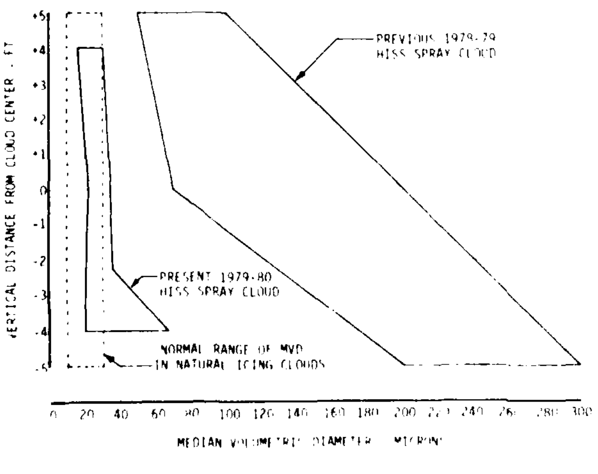

Fig. 14. Comparison of Cloud Drop Median Volumetric Diameters

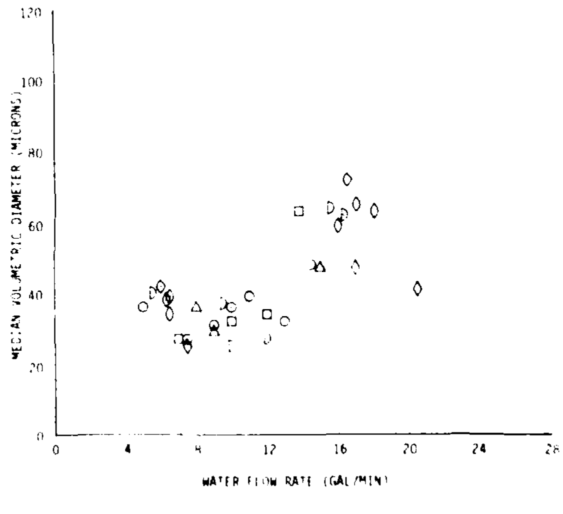

Fig. 15. Water Flow Rate and Median Volumetric Diameters of HISS Spray Cloud

VERTICAL VARIATION OF LIQUID WATER CONTENT WITHIN HISS SPRAY CLOUD

NOTES:
1. 97 SONICORE NOZZLES INSTALLED ON BOOM CENTER SECTIONS ONLY
2. 150 FOOT STANDOFF DISTANCE, 120 KTAS
3. CURVES BASED ON A COLLECTION OF VERTICAL SWEEP DATA FROM SEVERAL FLIGHTS
4. CURVES SHOWN DO NOT REPRESENT EXACT VALUES, BUT SUGGEST AVERAGE TRENDS BASED ON COMBINED DATA

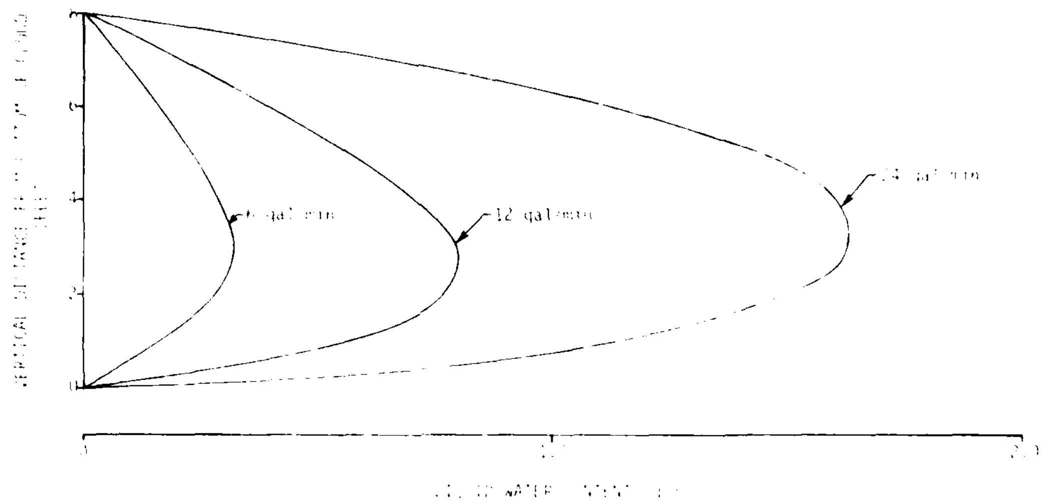

Fig. 16. Vertical Variation of Liquid Water Content Within HISS Spray Cloud

WATER FLOW RATE AND LIQUID WATER CONTENT OF HISS SPRAY CLOUD

SYMBOL	OUTSIDE AIR TEMPERATURE (°C)	PRESSURE ALTITUDE (FT)	RELATIVE HUMIDITY (%)
[illegible]	-10.0 to -13.0	6800 - 9900	10 - 13
[illegible]	- 5.0 to -15.0	3600 - 8900	24 - 29
[illegible]	- [illegible] to -19.5	3860 - 9960	55 - 57
□	-15.0 to -17.0	2800 - 9040	60 - 67
◇	-10.0 to -15.0	2500 - 8400	72 - 78
○	- 5.0 to -20.0	6000 - 9240	80 - 85
△	- 5.0 to -15.0	3840 - 9260	90 - 97

NOTES:
1. 97 SONICORE NOZZLES INSTALLED ON BOOM CENTER SECTIONS ONLY.
2. DATA SHOWN ARE FOR STABLE IMMERSIONS CENTERED IN SPRAY CLOUD.
3. 150 FT. STANDOFF DISTANCE, 120 KTAS

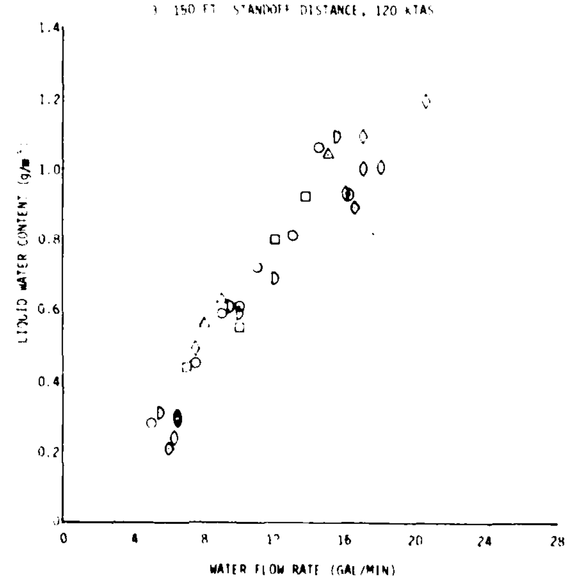

Fig. 17. Water Flow Rate and Liquid Water Content of Spray Cloud

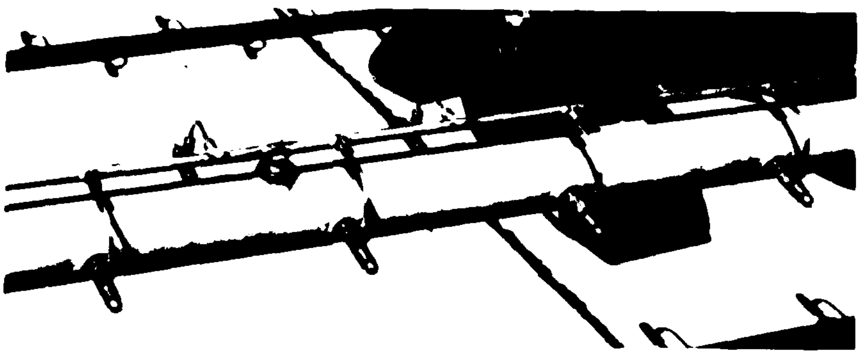

Fig. 18. HISS Boom Water Manifolds

AIRCRAFT CANDIDATE	USEFUL LOAD ~ LBS (Kg)	AIRSPEED RANGE ~ KTAS (Km/Hr)	ALTITUDE CAPABILITY ~ FT (m)	CABIN SIZE ~ FT (m)	COMMENTS
CH-47C (FRB)	24,162 (10,945)	0 - 150 (0 - 278)	15,000 (4,572)	30 x 7.5 x 6.5 (9.1 x 2.3 x 2.0)	Current HISS
CH-47D	26,873 (12,173)	0 - 150 (0 - 278)	15,000 (4,572)	30 x 7.5 x 6.5 (9.1 x 2.3 x 2.0)	Qualified to 50,000 lbs (22,650 kg)
CH-47D	30,873 (13,275)	0 - 140 (0 - 259)	12,000 (3,658)	30 x 7.5 x 6.5 (9.1 x 2.3 x 2.0)	Assumes qualification to 54,000 lbs (24,462 kg)
CL215	15,562 (7,050)	66 - 164 (122 - 304)	16,200 (4,938)	31 x 7.8 x 6.3 (9.4 x 2.4 x 1.9)	Piston Powered
C-130	81,382 (36,866)	100 - 325 (185 - 602)	33,000 (10,058)	40 x 10 x 9 (12.2 x 3 x 2.7)	Air Force PAWSS
KC-135	197,866 (89,633)	126 - 600 (233 - 1111)	42,000 (12,801)	112 x 12 x 7.5 (34 x 3.7 x 2.3)	Speed limited
CH-53D	18,515 (8,387)	0 - 170 (0 - 315)	21,000 (6,401)	30 x 7.5 x 6.5 (9.1 x 2.3 x 2.0)	Availability limited
CH-53E	35,750 (16,195)	0 - 170 (0 - 315)	18,500 (5,639)	30 x 7.5 x 6.5 (9.1 x 2.3 x 2.0)	Availability limited

Fig. 19. Candidate Aircraft Chart for HISS Air Vehicle

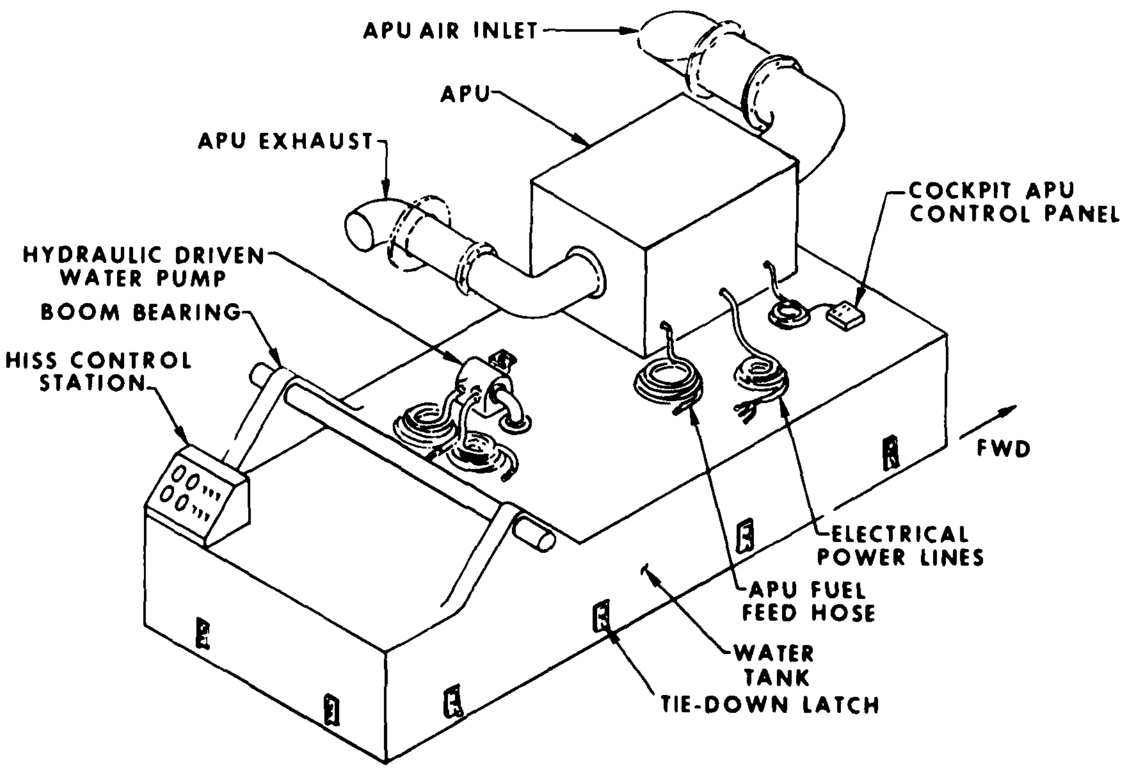

Fig. 20. Conceptual Design of Palletized HISS

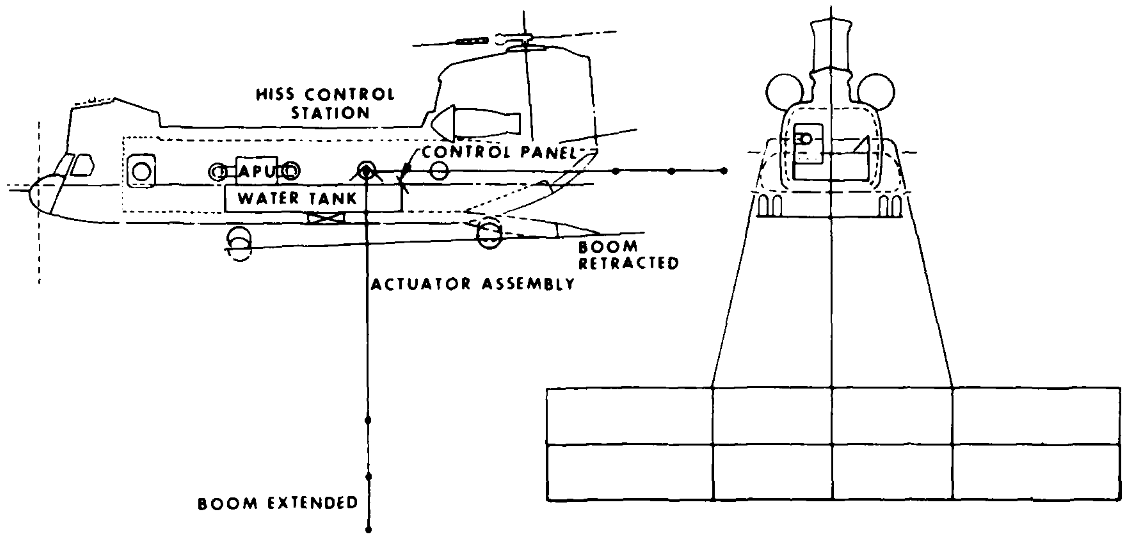

Fig. 21. Conceptual View of Selected CH-47C/D HISS Air Vehicle

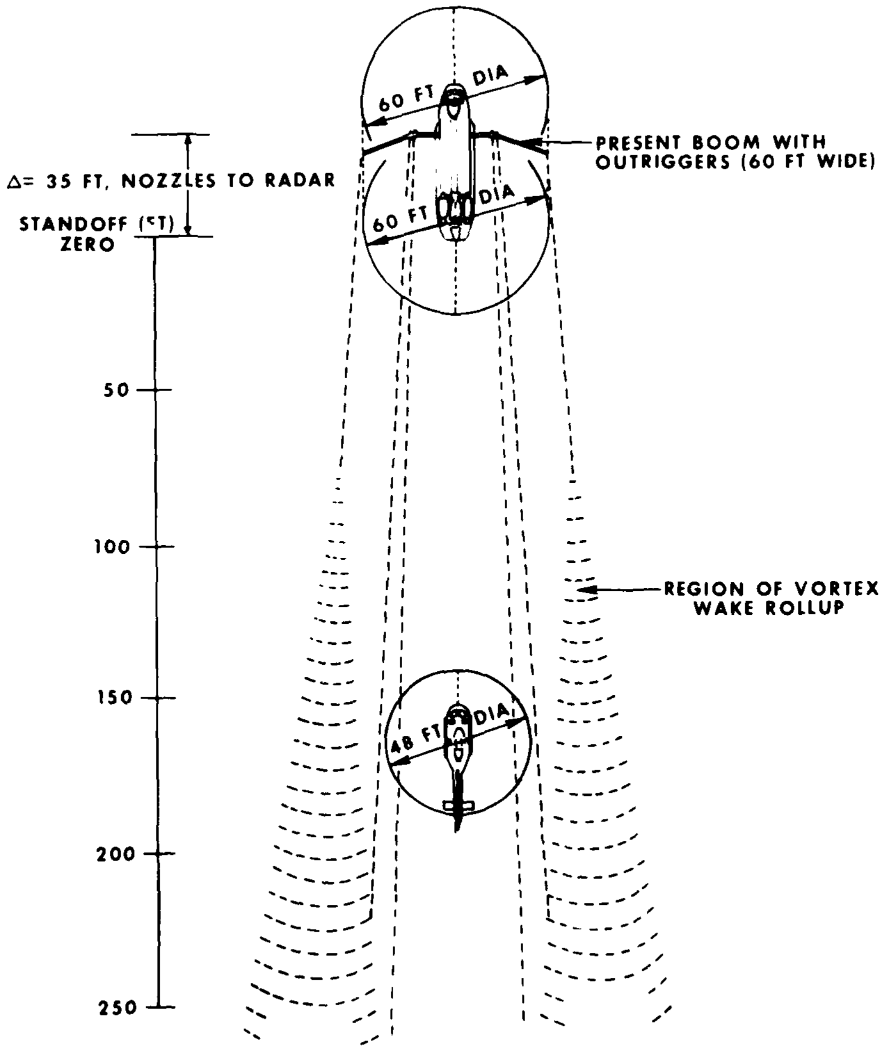

Fig. 22. HISS Cloud Spray Pattern Due to Vortex Wake Rollup

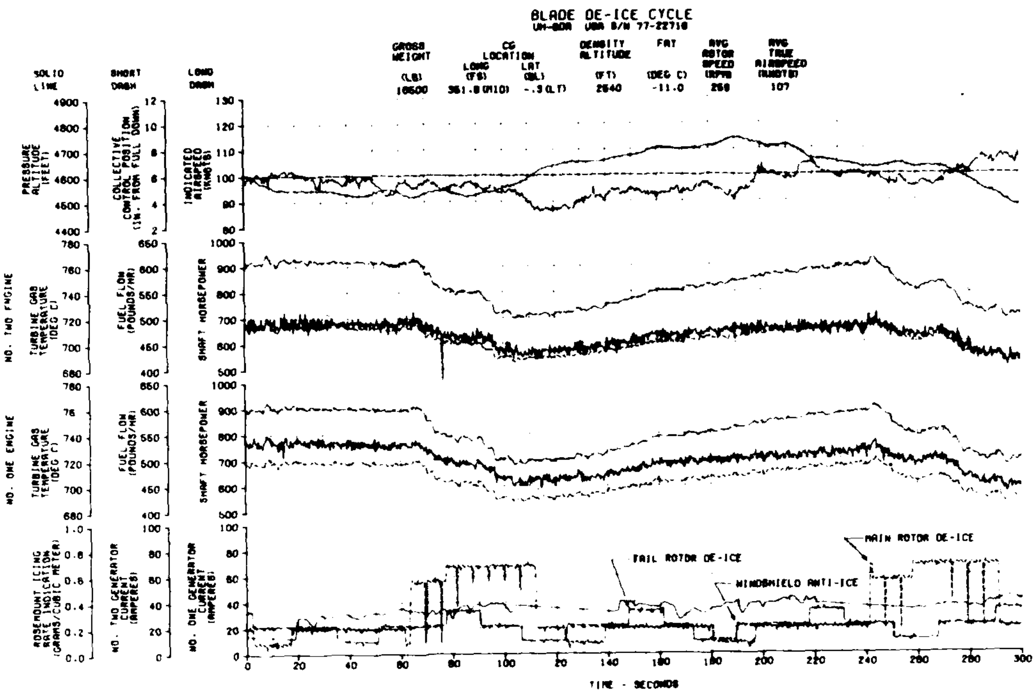

Fig. 23. Time History of UH-60A Increase in Power Required During Deice Cycle

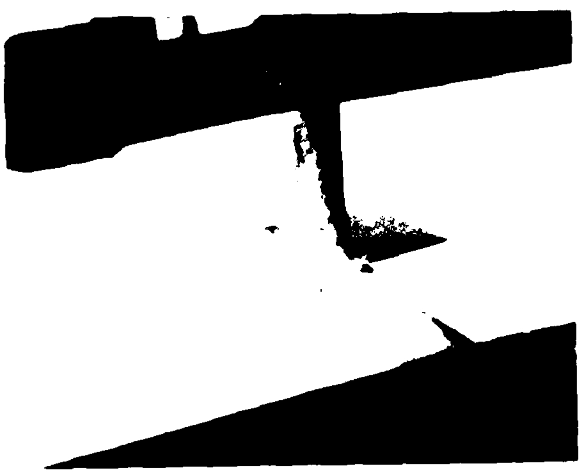

Fig. 24. Typical Natural Ice Horn Protuberances

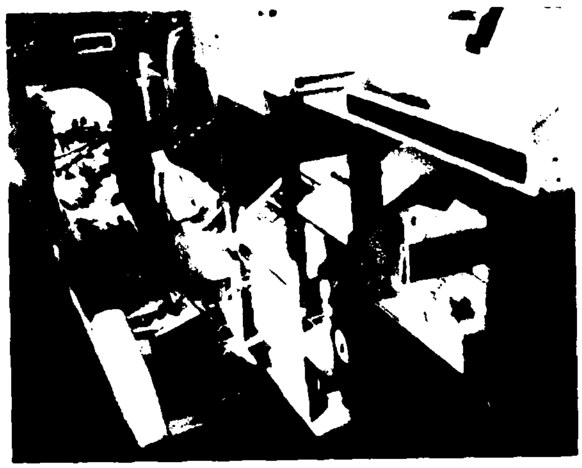

Fig. 25. U.S. Army U-21A Exterior Showing Cloud Measuring Equipment Installation

Fig. 26. U.S. Army U-21A Interior Showing Small Intelligent Ice Data System Installation

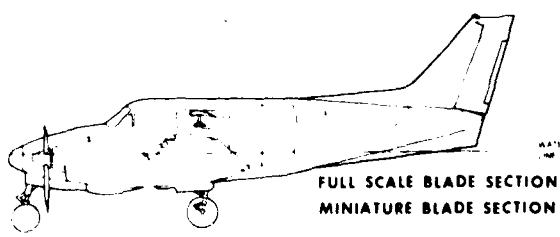

Fig. 27. Conceptual Design of Airfoil Sections on U-21A

AIRCRAFT OPERATIONS FROM AIRFIELDS WITH SPECIAL UNCONVENTIONAL CHARACTERISTICS

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1) INTRODUCTION

Devant le développement du transport type "commuter" et la mise en service à la fin des années 60 d'avions de masse inférieure à 5700kg mais d'une capacité allant jusqu'à 20 passagers les autorités de différents pays ont estimé devoir modifier les réglements de navigabilité et les réglements opérationnels pour assurer à ce type de transport un niveau de sécurité comparable à celui du transport aérien classique (SFAR 23 et appendix A de la FAR 135 aux Etats-Unis, Arrêté du 25 Novembre 1974 en FRANCE par exemple).
L'essentiel de ces nouvelles dispositions concerne l'exploitation des avions multimoteurs de masse inférieure à 5700kg aménagés pour le transport de 10 passagers et plus pour lesquels il faut couvrir la panne de moteur au décollage. Ceci implique :

- . De tenir compte de la distance d'accélération-arrêt
- . De limiter eventuellement la masse maximale au décollage afin d'assurer une trajectoire au 2e segment de pente minimale 2 p.cent avec un moteur hors de fonctionnement.

Deux types d'exploitations envisagées en FRANCE sur des pistes courtes dépourvues de prolongement d'arrêt étaient incompatibles avec la première exigence :

- . L'exploitation sur certains terrains Outre mer situés en général sur des îlots
- . L'exploitation sur altiports déneigés (cf figure 1) ou l'accélération arrêt est impossible dés que l'avion se trouve engagé dans la pente, qui peut atteindre - 20p.cent comme à COURCHEVEL.

Pour ces exploitations l'implantation de pistes sufisamment longues pour satisfaire les exigences de l'arrêté du 25 Novembre 1974 n'est pas possible, soit à cause du manque d'espace soit parce qu'il est préférable de privilégier d'autres facteurs favorables à la sécurité comme l'axe de piste par rapport aux vents dominants ou par rapport aux obstacles. Ces pistes sont par contre pourvues d'excellents prolongements dégagés. Pour continuer à exploiter sur ces terrains des dérogations étaient nécessaires. Elles devaient être justifiées par des conditions techniques d'exploitation garantissant un niveau de sécurité satisfaisant. Pour que ce niveau de sécurité reste équivalent à celui visé par l'arrêté du 25 Novembre 1974 il restait nécessaire d'établir de nouvelles régles couvrant la panne d'un moteur au décollage. Il fallait en outre définir de nouvelles procédures et s'assurer que les exploitants avaient les structures adequates pour les appliquer. Ce travail, coordoné par la DGAC (Direction Générale de l'Aviation Civile), a été éffectué en coopération avec le CEV (Centre d'essais en vol) l'OCV (Organisme du Contrôle en vol de la DGAC) et la compagnie AIR ALPES. La présente communication rend compte des étapes techniques qui ont été nécessaires pour élaborer ces nouvelles régles.

2) HISTORIQUE

- 11 Août 1962 : Arrêté définissant les conditions techniques d'emploi des avions de transport public.
- 25 Novembre 1974 : Arêté définissant les nouvelles conditions techniques relatives aux performances exigées pour l'exploitation des avions de transport public.
- 17 Décembre 1975 : Autorisation d'exploitation par AIR ALPES des altiports de COURCHEVEL, MEGEVE.
- 16 au 26 Octobre 1978 : Essais en vol sur pistes plates ou légérement en pente (MENDE = 4%) avec un DHC6-300 : 29h de vol.
- 17 au 27 Spetembre 1979 : Essais en vol sur altiports avec un DHC6-300 : 19h de vol 70 décollages.

Nota : 7 pilotes (3CEV, 2DGAC, 2 AIR ALPES) ont participé aux essais en vol ci-dessus.

- 78/79 Mise au point d'un modèle de calcul des performances de roulement du DHC6-300 sur un calculateur IRIS 80 de la DGAC.
- Juillet 1980 - Rapport d'essais en vol.
- 18 Août 1982 - Autorisation d'exploitation du DHC6-300 sur pistes courtes par AIR CALEDONIE.
- 17 Novembre 1982 - Autorisation d'exploitation du DHC6-300 sur pistes courtes par AIR GUADELOUPE.
- 18 Décembre 1982 - Autorisation d'exploitation du DHC6-300 sur altiports, conformément aux recommandations ci-dessous, par TAT.

3) PROBLEME POSE PAR LES EXIGENCES AU DECOLLAGE (cf Figure 2)

Pour un avion de masse 5700Kg aménagé pour transporter 10 passagers ou plus la longueur de piste nécessaire doit être telle que :

L roulement (AEO) jusqu'à Lift off $\leqslant$ L runway
L 50ft (AEO) $\leqslant$ L runway + clearway
LASD (Acceleration stop distance) $\leqslant$ L runway + stopway avec V1/VLOF $\geqslant$ 1,1 VMCA ou 1,1VS (le plus grand) et V2 $\geqslant$ 1,1VMCA ou 1,2VS (le plus grand)

Exemple pour un DHC6-300 : m = 5670kg ISA + 25° Z = 0 sans vent et sur piste plate.

Llof (AEO) = 350m	V1 = VLOF = 74,5Kt	
L50ft (AEO) = 548m	V2 = 81 Kt	
LASD = 786m		

Dans ces conditions la pente au deuxième segment avec un moteur en panne (OEI) = 2,8% (Cette valeur montrant au passage que le DHC6-300 n'est pratiquement jamais limité poids-pente).
Il est clair que sur un terrain sans stopway la condition qui dimensionne la longueur de piste nécessaire au décollage est LASD.
On peut faire au passage trois remarques :
-1- La réglementation n'impose pas de condition piste associée au décollage avec un moteur en panne. Ceci se justifie pas le choix d'une V1 élevée. Dans notre exemple la distance jusqu'au LOF avec panne moteur à VEF = 72,5kt (soit 1" avant V1) serait de 510m soit nettement inférieure à la longueur exigée par la condition Accélération-Arrêt 786m.
-2- Pour l'équipage la procédure est simple car basée sur la connaissance de deux vitesses V1 et V2 ce qui revient à faire la rotation à V1.
-3- Pour le postulant les essais en vol de certification sont simplifiés par rapport aux avions certifiés FAR 25.
. Les mesures des distances de décollage OEI ne sont pas exigées
. Il n'y a pas de conditions associées au choix de VR (cas abusifs, essais de VMU)
. La mesure de VMCG (minimum control speed ground) n'est pas exigée.
l'exemple précédent montre clairement que pour décoller sur une piste plus courte, tout en assurant l'accélération arrêt en cas de panne moteur avant V1, il faut choisir une vitesse de décision plus faible, introduire la notion de VR vitesse de rotation au décollage qui sera supérieure à V1 et balancer la piste entre la distance de décollage et la distance d'accélération arrêt, ce qui n'était pas le cas précédemment. La distance de décollage à considérer doit être celle avec un moteur en panne.

4) PROCEDURE PROPOSEE SUR PISTE PLATE (cf figure 3)

Des discussions préalables entre la DGAC (SFACT et OCV) et le CEV, il est ressorti que la procédure de décollage devait être basée sur les hypothèses suivantes :
(1) Ne considérer comme devant être égale à la longueur de piste que la phase roulement au sol du décollage. La partie en vol, allant de l'endroit où l'avion quitte le sol (Point de LOF = lift off) jusqu'au franchissement des 50ft réglementaires, étant alors couverte par le prolongement dégagé d'obstacle que constitue par hypothèse le terrain concerné par la dérogation.
(2) Définir une vitesse de panne moteur (VEF = engine failure) inférieure à VR telle que la longueur de piste permette :
- soit d'accélérer tous moteurs en fonctionnement jusqu'à VEF, puis avec un moteur en panne de VEF à VLOF (Vitesse de décollage).
- Soit d'accélérer, tous moteurs en fonctionnement jusqu'à VEF puis d'arrêter l'avion dès reconnaissance de la panne (avec un Δt = 1" entre EF et l'action sur les freins).
(3) Pour pouvoir poursuivre la course au décollage en sécurité, il est nécessaire de pouvoir garder le contrôle au sol avec un moteur en panne à VEF. La détermination de la vitesse minimale de contrôle au sol, VMCG, n'est pas exigée lors des certifications actuelles SFAR 23, mais l'est par contre en FAR 25 c'est à dire pour les avions de plus de 5 700kg et est démontrée en utilisant les seules gouvernes aérodynamiques. La démonstration peut donc être effectuée par un seul pilote. Il est donc proposé de prendre les mêmes démonstrations et exigences, c'est à dire VEF $\geqslant$ VMCG (gouvernes aérodynamiques).
(4) Garder, en les justifiant par les essais les vitesses de décollage prévues dans l'arrêté soit :
VLOF $\geqslant$ 1,1 VMCA ou 1,1 Vs.
V2 $\geqslant$ 1,1 VMCA ou 1,2 Vs.
(5) Garder les limitations poids-pente actuelles (0% au premier segment, 2% au deuxième segment).
Les questions posées par la proposition précédente sont :
- Détermination de VMCG
- Détermination des performances de roulement avec un moteur en panne
- Justification du choix de VR/VLOF
- Marges à prendre au roulement pour tenir compte de la dispersion opérationnelle.
- Définition du clearway pour couvrir la distance de décollage un moteur en panne jusqu'à 50ft.

5) CAS PARTICULIER DES ALTIPORTS (cf figure 4)

5.1 - La décision
Sur altiports l'extrême rapidité du déroulement des séquences de décollage et la faible probabilité de pouvoir arrêter l'avion une fois qu'il est engagé dans la pente rendent inadaptée la notion classique de V1. A COURCHEVEL par exemple l'avion est déjà à 35Kt dans la pente à -20%. Laisser, à cet instant, le choix au pilote entre la continuation du décollage ou son interruption risque de conduire à un freinage tardif et donc à une sortie de piste. Pour éviter toute confusion l'équipage doit être programmé pour le décollage dès le passage de la cassure de pente - et à la notion de vitesse de décision il faut substituer la notion de point de décision. (Des essais en vol éffectués avec un Breguet 941 suggéraient même de prendre V1 = 0 :cf référence 1). Sur la pluspart des bimoteurs actuels il est illusoire d'espérer une VMCG, démontrée avec les seules gouvernes aérodynamiques, inférieure à 50Kt. Une possibilité d'éviter la perte de contrôle entre le point de décision ($\simeq$ 30Kt à COURCHEVEL) et la VMCG est d'autoriser, réglementairement, l'utilisation de la dirigeabilité pour garder le contrôle de l'avion au sol.
5.2 - Distance de décollage et Pente de montée après décollage (1er et 2eme segment)
Les altiports sont en général assez hauts, 6800ft pour COURCHEVEL, et situés au dessus de vallées dégagées et en cas de panne moteur le terrain de dégagement est situé dans la plaine. Ces facteurs rendent inadéquats les critères habituels utilisés sur piste plate : La distance de décollage à 50ft n'a pas de sens operationnel. Les exigences de montée au 2eme segment (2%) avec un moteur en panne peuvent être inutilement pénalisantes.

5.3 - Congères de neige
En hiver les altiports sont déneigés mais il peut substituer en bout de piste des congères pouvant atteindre 1m.

5.4 - Dérogation sur altiports (cf. figure 5)
Des considérations précèdentes, il ressort que la procédure de décollage doit être basée sur les hypothèses suivantes :

(1) Définir un point de décision situé à la cassure de pente et tel que la longueur de piste permette d'accélérer tous moteurs en fonctionnement jusqu'à VEF puis avec un moteur en panne de VEF à VLOF et de passer le seuil de piste avec une marge suffisante (> 1 m si neige)

(2) Poursuivre le décollage en sécurité en démontrant que l'avion est controlable à partir de VEF en utilisant les gouvernes aérodynamiques et/ou la dirigeabilité
VEF ≥ VMCG (DIRIGEABILITE)

(3) Garder les vitesses de décollage prévues par l'arrêté soit :
VLOF ≥ 1,1 VMCA ou 1,1VS
V2 ≥ 1,1 VMCA ou 1,2VS

(4) Ne pas réglementer à priori les trajectoires de décollage et les pentes de montée après le passage du seuil avant d'avoir analysé les résultats d'essais.

Les questions posées étant :
- Détermination de VMCG avec dirigeabilité
- Détermination des performances de roulement avec un moteur en panne
- Justification de VR/VLOF
- Marges à prendre au roulement pour passer le seuil de piste à une hauteur suffisante
- Propositions de critères au décollage après le seuil de piste

6) APPLICATION DE LA DEROGATION AU DHC6-300

Afin de vérifier les procèdures spéciales proposées, en connaître les limites et répondre aux questions nouvelles posées, des essais en vol avec un avion représentatif instrumenté étaient nécessaires. Il a été admis avec la DGAC de limiter dans un premier temps cette procèdure au DHC6-300 pour plusieurs raisons :

- Cet appareil est l'avion le plus répandu pour l'exploitation outremer du fait de ses caractéristiques rustiques et de ses bonnes performances qui ont fait qu'il était adapté aux conditions locales d'utilisation d'une infrastructure réduite lorsque la réglementation en vigueur était celle de l'arrêté du 2 Août 1962.
- Le DHC6-300 est un avion théoriquement bien connu du fait des études menées par le bureau "Technique utilisation" pour l'utilisation des altiports.
- Une recherche des caractéristiques et des performances détaillées de l'appareil, des contacts avec de HAVILLAND of CANADA, ont permis d'élaborer des programmes sur HP 65 et IRIS 80 couvrant certaines performances de l'avion en fonction de plusieurs paramètres du domaine du vol et capable de fournir les données nécessaires à l'étude présente.

6.1 - Caractéristiques du DHC6-300
Cet avion étant déjà bien connu, seules les caractéristiques particulières intéressant l'expérimentation sont rappelées ci-dessous

Limitations
Masse maximale au décollage 5 670 kg
Masse maximale à l'atterrissage 5 580 kg
VMCA Volets 10° = 66 kt
Puissance maximale Décollage = 620 HP
et Maxi Continu
Couple maximum = 50 psi

Volets : la configuration de décollage est normalement 10°, la configuration d'atterrissage est normalement 37°5 mais il est possible d'afficher des positions de volets intermédiaires.

Le train est fixe :
- Ce qui a facilité l'implantation et le cheminement de l'installation d'essais
- A même performance au 2ème segment, la performance au décollage d'un avion à train fixe est donc meilleure ce qui est favorable à la sécurité des essais.

Performance :
L'avion est motorisé par deux turbopropulseurs à turbine libre PT6-A27 de 620 HP. Il s'agit de moteurs détarés pour lesquels la puissance est d'abord limitée par l'arbre de sortie et non par la puissance thermique de la turbine motrice. Ceci permet de conserver la puissance nominale jusqu'à des températures et altitudes élevées ce qui est particulièrement intéressant outremer et en montagne.
D'une manière générale la performance est brillante. On peut, par exemple, encore décoller la masse maximale à Z = 6000 ft et Ts = 19°C (standard + 16°C)

Dirigeabilité :
L'avion est équipé d'une dirigeabilité qui est commandée, hydrauliquement, par un levier situé sur le volant de la place gauche, ce qui permet un transfert dirigeabilité - gauchissement rapide. Il n'est néanmoins pas possible à un seul pilote d'utiliser simultanément les deux commandes.

Instruments moteurs :
Le couple est le paramètre opérationnel que le pilote doit afficher, en fonction des conditions Zp et Ts du jour, pour observer les limitations moteurs : couple max, T5 max, NG max...
La puissance développée sur l'arbre de sortie est donnée par

$$P = 12,88 \times \text{Torque (psi)} \times \frac{Np\ \%}{100} \qquad Np = 96\ \%\ \text{au décollage}$$

Drapeau automatique :
Chaque hélice est équipée d'un dispositif de mise en drapeau automatique.
Simulation des performances de montée :
Les performances peuvent facilement être simulées à partir du paramètre de conduite : le couple. Cete simulation permettait de balayer, à titre d'étude générale, la plage des conditions poids-pente prévisibles pour les avions susceptibles d'être concernés par la dérogation.
Simulation de la panne moteur :
La turbine et l'hélice étant libre, on peut simuler une panne moteur en passant l'hélice en drapeau tout en conservant le générateur en fonctionnement.

6.2 - MOYENS DE MESURES

L'installation d'essais réalisée a été la suivante :

6.2.1 - Mesures sur l'avion
- Enregistrées sur HB A 1322 (PV 6 mm/s : GV 30mm/s).

Paramètres enregistrés sur HB	Comment	Objectif
Δp et pr	Circuit Co-Pilote	Mesure de Vc
Vitesses de roues	Une génératrice tachymétrique sur chaque roue principale	Mesure de Vsol et datation du moment de décollage LOF
Compte tours de roue	Détecteur de proximité et lance top sur chaque roue principale	Déterminer la distance de roulement par mesure interne (intégration des tops)
Jx	Accéléromètre sur plancher	Dater la panne moteur
Boîtier X 2018	VHF2 circuit Copilote	Synchronisation des cinéthéodolites
Tops photos Caméra		Synchronisation

- Photo Panneau pour les paramètres moteurs.

6.2.2 - Moyens sol
Trajectographie :
- Par un cinéthéodolite, pour étalonner les génératrices tachymétriques et sur altiports pour avoir la trajectoire de décollage.
- Repères sol sur la piste et observations visuelles de la distance de roulement sur altiports.

Une station météo en piste : Pour mesurer le vent, la pression et la température.
Une écoute radio : Sur altiport pour assurer la sécurité et transmettre les observations visuelles.

7) RESULTATS GENERAUX

7.1 - Simulation de la pente au deuxième segment
Nous voulions effectuer des essais en jouant le jeu operationnel aux différentes pentes potentielles W/V (ou limites poids-pente) suivantes

W/V = 3,5 % Valeur représentative de la pente au 2e segment un moteur en panne d'un DHC6-300 pour les exploitations visées
W/V = 2 % Valeur représentative de la pente minimum réglementaire dans les mêmes conditions
W/V = 0 Valeur représentative d'un avion limité poids-pente au 1er segment par la réglementation actuelle - et intéressante à analyser sur altiports.

Pour des raisons de sécurité il était de plus nécessaire pour les altiports de simuler cette pente potentielle avec une puissance symétrique. La méthode suivante basée sur l'affichage à priori du couple = f(mg, W/V) a donné satisfaction et peut être préconisée pour l'entrainement et les décollages à puissance détarée.

$$W/V = \frac{TN}{mg} - \frac{1}{f}$$

- $1/f$ ne dépend pratiquement que de la configuration et des kVs de vol qui étaient toujours les mêmes.

- TN traction nette $= \eta \frac{P}{V} = \eta \times \frac{12,88}{V} \times \text{Torque} \times Np$ | P = puissance moteur | η = rendement hélice

La figure 6 montre que le rendement de l'hélice ne dépend pas de la puissance aux vitesses de décollage. En monomoteur réelle Np est constant et égal à 96 % au décollage. (dès 40 Kt environ). On peut donc simuler le W/V a iso Torque le Torque étant affiché à iso vitesse.
La figure 8 confirme que la performance peut être déterminée à partir de la seule connaissance du torque. Sur cette planche en effet toutes les conditions (Zp, Ts) situées sur une horizontale conduisent à afficher le même torque à 73 Kt.
L'axe des ordonnées peut donc être gradué en Torque.
La figure 7 établie à partir des données précédentes, confirme qu'à un W/V visé corespond une valeur de Torque/mg pratiquement indépendante de la masse.
Ce qui précède est valable pour une simulation en monomoteur (Np = 96 %). Pour une simulation en puissance symétrique il faudrait (?) afficher symétriquement la moitié du torque ainsi déterminé. En fait, deux difficultés surgissent : Le régime hélice est inférieur à 96 % aux faibles torques et il faut simuler la traînée dissymétrique qui est équivalente à une baisse de traction de 10 %. Le torque à afficher symétriquement sur les deux moteurs est donc : Torque(2)= 1/2x 0,9 Torque(1)x 96 % / N2
L'analyse de l'évolution du régime hélice au décollage en fonction de la vitesse et du torque (cf figure 9) permet de tracer sur la figure 11 les torques à afficher en puissance symétrique.
Le calcul à posteriori des W/V réels a montré que cette méthode donnait satisfaction.

7.2 - Calcul des performances de roulement

L'équation du roulement au sol est :

$$\Gamma/g = \frac{TN}{mg} - \mu - (CD - \mu CL)\frac{qS}{mg} - \pi$$

avec
TN = Traction nette = f (V, Torque, Np, σ)
μ = coefficient de roulement (0,025 sur piste en dure)
π = pente de la piste : modelisée f(L) pour chaque piste
σ = densité de l'air f (Ts, Ps)
$q = 1/2\, \rho\, V^2$ (pression dynamique)
V = f (Δp, σ)
Vsol + vent
CD/CL = coefficients de traînée/portance à valider

Les conditions d'essais (qui sont les variables opérationnelles m, μ , vent, profil de piste, point de lâcher des freins, σ) sont introduites en données. Les résultats du calcul L = f (Vsol) sont alors comparés aux résultats d'essais. Les bases peuvent être modifiées pour minimiser les écarts et, l'objectif final qui est de valider le modèle théorique peut alors être atteint rapidement.
Cette technique est la plus efficace dès lors qu'on a un minimum de données du Constructeur. Elle a été retenue pour le DHC6-300, suite aux contacts fructueux entre la DGAC et DHC et le modèle a été programmè sur l'ordinateur IRIS 80 de la DGAC.
Pour valider au mieux les bases de calcul, il était important d'avoir la meilleure approximation possible de la traction nette.

$$TN = \eta \frac{P}{V}$$

- avec η rendement de l'hélice obtenue par la "Propeller thrust chart" du constructeur. (cf figure 6). On a admis que le rendement d'hélice était le même aux régimes hélices pratiqués (entre 1900 et 2112 tours/mn).
- P = 12,88 X Torque X Np (cf figure 9 et 10 pour les valeurs de Np en fonction du torque)

A partir du réseau de données précédent établi pour chaque moteur en essai on obtient donc la traction nette à partir des données enregistrées (Torque, vitesse, σ)
La mise au point du modèle de calcul a ainsi pu être faite assez facilement en comparant les distances essais-calcul à iso Vsol. Ce modèle simple rend bien compte des performances de roulement du DHC6-300 et non des performances de décollage en effet de sol

7.3 - VMCA

On a confirmé la valeur publiée de 66 kt avec volets 10°. On a recherché, en statique, la VMCa avec volets 20° ; elle est légèrement inférieure, environ 65 kt , ce qui correspond comme vérifié ultérieurement, au chiffre annoncé par le Constructeur pour cette configuration.
La VMCa ailes horizontales se situe vers 73 kt.
On a donc :

VMCA	(ϕ = 5°)	Volets 10°	66 kt en Vc
		Volets 20°	65 kt en Vc
VMCA	(ϕ = 0°)		73 kt en Vc

En respectant la réglementation associée a VMCA les vitesses minimales au décollage sont les suivantes :

	Règlementation	Volets 10°	Volets 20°
VLOF min	≥ 1,1 VMCA	73 kt CAS	5 71 kt CAS
V2 min	≥ 1,1 VMCA		

7.4 - Performances de roulement volets 20°

La question ayant été soulevée de savoir s'il était intéressant, pour diminuer la distance lâcher des freins - LOF, de décoller avec les volets à 20°, nous avons effectué sur piste plate et sur altiports quelques mesures dans cette configuration.

L'influence des volets sur la performance de roulement dépend de la pente de la piste et de la motorisation.
A iso Vsol on trouve :
- sur piste plate = une augmentation de la longueur de 20 % en bi-moteurs.
  une augmentation de la longueur de 50 % en monomoteur.
- sur altiports = une augmentation de la longueur de 10 % quelque soit la motorisation.

Si la VLOF avait lieu à iso incidence (à 1,1 Vs par exemple de chaque configuration) on aurait, en prenant la configuration volets 20° au décollage, un gain, dû à une VLOF plus faible, de 15 %. Dans cette hypothèse la configuration volets 20° serait légèrement avantageuse sur altiports uniquement.
En fait, les valeurs de VLOF sont limitées par 1,1 VMCA, ce qui donne les "gains" suivants sur altiports.

m (lb)	VLOF (kt) Volets 10°	VLOF (kt) Volets 20°	Gain dû à VLOF plus faible	Perte due à traînée plus grande	Lr Volets 20°
12 500	75 (1,1 Vs)	5 71 (1,1 VMCA)	- 9 %	+ 10 %	idem
12000	73 (1,1 VMCA)	5 71	- 4 %	+ 10 %	Augmentée 6 %

On voit donc qu'il n'y a, à priori, aucun avantage à prendre la configuration volets 20° au décollage en ce qui concerne la performance de roulement.
Comme le contrôle au sol, en cas de panne moteur, est plus délicat avec volets 20° que volets 10°. Il est recommandé de ne pas autoriser cette configuration de décollage en utilisation normale.
Nota : Pour une utilisation spéciale où il serait justifié de faire l'impasse de la panne moteur, l'utilisation de la configuration Volets 20°, serait avantageuse sur altiports seulement.

8) RESULTATS SUR PISTES PLATES

8.1 - VMCG (gouvernes aérodynamiques seules)
Cette vitesse a été déterminée de façon classique c'est à dire en utilisant les gouvernes aérodynamiques seules, en affichant la puissance maximale et sans solliciter le manche en secteur avant. Plusieurs essais ont été effectués par vent de travers pour déterminer l'influence du vent. Les résultats sont donnés figure 12.
Les quelques points effectués avec volets 20° n'ont pas fait apparaître de différence décelable sur les résultats.
La valeur de VMCG par vent nul est égale à 56 kt et a été confirmée sur piste mouillée.
Le vent de travers augmente cete valeur de façon sensiblement égale à la moitié de la composante de vent traversier. Hormis cela le comportement avion à la panne n'est pas modifié.
Sur le plan opérationnel, il peut s'avérer utopique de rentrer systématiquement dans un abaque pour déterminer V1 min en fonction du vent. On pourrait préconiser V1 min = 61 kt, qui couvrirait les vents de travers jusqu'à 10 kt, et alerter l'équipage pour lui faire recalculer sa V1 min lorsque le vent de travers est supérieur à 10 kt.

8.2 - Choix de VR/VLOF

L'objectif des essais était de valider Vlofmin = 1,1Vs avec une vitesse de décision nettement inférieure.
Cette procédure change assez sensiblement le décollage monomoteur sur piste plate.
Dans le cas de la procédure réglementaire, la panne a lieu à la rotation et comme la traction ne chute pas instantanément on accélère assez vite vers V2.
Au contraire avec cette nouvelle procédure, à VR on est en monomoteur établi, l'accélération est beaucoup plus lente et dépend du Tn/mg, et l'on peut rester un long moment, près du sol quasiment en palier avant d'atteindre V2 lors les décollages sur pistes plates.
Dans les conditions de l'essai, à savoir turbulence faible et bonnes références extérieures, la procédure est aisément praticable même au Tn/mg mini avec les réserves ci-après :

1) Le contrôle très fin de l'assiette et le palier après VLOF pourra être pénible à exécuter sans bonnes références extérieures, par exemple de nuit ou au-dessus de la mer, car la "négociation" du compromis vario-accélération peut s'y avérer difficile à piloter au T/mg mini.
Opérationnellement on pourra jouer sur les deux limitations suivantes pour garder une procédure sûre :
  - Imposer une visibilité et autoriser les décollages au T/mg minimal.
  - Avoir un T/mg assurant un taux de montée confortable par exemple une pente de 3,5 % au 2eme segment, lorsque les références extérieures sont dégradées.

2) Il est recommandé pour ne pas être trop absorbé par les problèmes de contrôle transversal qui sont plus crucials sur piste étroite de prendre VLOF $\geqslant$ VMCA ailes horizontales. Dans le cas du Twin otter ce résultat est atteint automatiquement en prenant VLOF = 1,1 VMCA puisque

$$1{,}1\ \text{VMCA}\ (\Phi = 5°) = \text{VMCA}\ (\Phi = 0) = 73\ \text{kt}.$$

Les essais confirment donc qu'il n'est pas envisageable de déroger à l'exigence VLOF $\geqslant$ 1,1 VMCA.

3) Le décollage par fort vent arrière à VLOF = 1,1 Vs en monomoteur établi est déconseillé à cause des eventualités de gradient de vent défavorable.

Compte tenu de la faible accélération en monomoteur établi, on recommande d'associer à VR les limitations choisies à VLOF.

8.3 - Distance de décollage sur piste plate avec un moteur en panne

Bien que la réglementation en vigueur n'exige pas cette mesure les informations données par DHC donnent une idée de ce que serait cette distance avec la V1 réglementaire pour l'exemple déjà choisi (cf figure 13).
Bien qu'il soit acceptable de décoller à une VLOF égale à 1,1 Vs, il reste admis que la vitesse de sécurité au décollage V2 doit être conservée égale à 1,2 Vs. Il faudra donc accélérer près du sol. Les clearways des terrains sur lesquels la procédure spéciale sera autorisée devront donc avoir une longueur supérieure à la distance réglementaire actuelle séparant le décollage LOF du point ou l'avion atteint V2 et 50ft tous moteurs en fonctionnement.
En vue d'évaluer cette distance quelques trajectoires ont été suivies aux cinéthéodolithes pour des W compris entre 3,5 % et 2 % la distance air parcourue a été comprise entre 500m et $\overline{\text{V}}$ 1000 m . On retiendra donc ce dernier chiffre pour fixer les contraintes liées au clearway.
Les valeurs réelles (fonction de la pente au 2eme segment) peuvent être obtenues auprès de DHC. A défaut un calcul conservatif qui ne prend pas en compte l'effet du sol, pourra être appliqué (cf figure 14)

8-4 Dispersion operationnelle

Nous n'avons pas joué le jeu de la mesure de la dispersion opérationnelle sur pistes plates, ce jeu ne pouvant être valable qu'en conditions réelles (limite poids-piste).
Des avis pilotes, on peut néanmoins dégager l'ordre de grandeur de la marge minimale à prendre au roulement pour avoir encore la vue du bout de piste et tolérer un délai de réaction d'une seconde à VR.
Cette marge devrait être d'environ 50 m (ou encore en étant plus conservatif, proche de 10 % de L piste).
Il est évident que cette marge peut être adaptée sur chaque terrain en fonction de la nature de "l'overrun".

8.5 Conclusions et recommandations sur piste plate

En appelant "procédure spéciale" toute procédure TPP2 ou V1 serait inférieure à 1,1 Vs ou 1,1 VMCA nous recommandons la procédure suivante.

Clearway		1000 m
Volets décollage		10°
Vent arrière	<	10 Kt
V1 min	=	61 Kt si vent travers $\leqslant$ 10 Kt
	=	$61 + \frac{W-10}{2}$ si vent travers > 10 Kt

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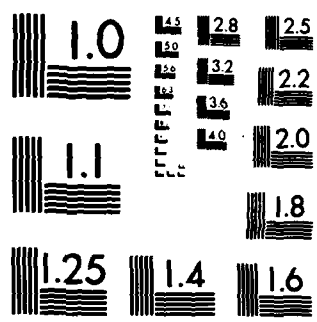

MICROCOPY RESOLUTION TEST CHART
NATIONAL BUREAU OF STANDARDS-1963-A

<u>Masse maximale autorisée</u>
<u>Limitations poids-pente</u> inchangées *
LASD (panne moteur à VEF) $\leqslant$ L runway
LLOF (panne moteur à VEF) $\leqslant$ L runway-50m
*Si la visibilité est dégradée il est recommandé de prendre une limite poids-pente monomoteur = 3,5 % au 2eme segment.

<u>Procédure</u> V1 determiné par la relation précédente
VRmin = 1,1 Vs ou 1,1 VMCA
V2 = 1,1 Vs ou 1,2 VMCA

<u>Equipage</u> Un pilote qualifié

9) RESULTATS SUR ALTIPORTS

9.1 - VMCG en utilisant la diregeabilité

On a vérifié qu'en cas de panne moteur la dirigeabilité permettait de conserver l'axe avec un faible écart initial (3m). Le résultat a été vérifié en monomoteur réel, jusqu'à des vitesses de panne de 25kt avec vent de travers ($\simeq$ 10kt) sur piste plate légèrement mouillée et en monomoteur simulé à une vitesse de panne d'environ 30 kt sur les altiports.

Au sol, le contrôle transversal à la panne moteur est assuré par la combinaison d'un braquage de la direction et de la dirigeabilité qui vient en complément, afin de conserver l'axe à une vitesse inférieure à VMCg aérodynamique.
Une certaine coordination équipage est nécessaire, en particulier dans la zône des 50/60 kt où la profondeur doit être maintenue secteur avant, afin d'assurer une bonne adhérence de la roulette de nez.
De plus, la dirigeabilité ne doit pas être trop braquée, évitant ainsi de faire déraper le pneu.
A l'envol il est nécessaire, compte tenu de la proximité de VMCa, de braquer la direction proche de la butée, voire d'incliner légèrement l'avion pour assurer le contrôle de cap.
La procédure proposée qui nécessite un équipage à deux pilotes est acceptable sous la forme suivante :

- le pilote place gauche met les gaz sur freins et tient la dirigeabilité de la main gauche
- le copilote maintient le manche secteur avant, ce qui a pour conséquence de maintenir la roue avant en contact avec le sol, gauchissement au neutre, et annonce les vitesses à partir de 30 kt (minimum lisible), de 10 kt en 10 kt jusqu'à VR.
- à VR le pilote lache la dirigeabilité, prend le volant en main, et décolle (rappelons que la manette de dirigeabilité est située environ à 10 cm du volant, sur la colonne du manche, et que le passage de l'une à l'autre est facile).

Le pilote n'ayant donc le contrôle du gauchissement qu'après VR, cette procédure limite la composante travers de vent admissible à une valeur faible inférieure à 10 Kt.
Enfin, les pilotes qualifiés Altiport devront subir un entrainement (sur piste horizontale) avec panne par réduction et drapeau.

9.2 - Choix de VR/VLOF

Sur altiports, il n'y a aucun problème à décoller à VLOF = 1,1 Vs compte tenu de l'accélération nettement plus élevée de l'avion. Il est même possible de prendre VR inférieur à VLOF pour tenir compte de cette accélération. Compte tenu du grand nombre de décollages effectués par plusieurs pilotes, les résultats statistiques amènent à recommander de prendre VR = VLOF - 3 kt

9.3 - Trajectoires au décollage : marge au roulement

Pour un décollage limite poids-piste, il semble que le pilote est sensible au temps le séparant du bout de piste et qu'il entame naturellement la rotation environ 1 seconde avant d'atteindre le bout de piste.
L'équipage souhaite avoir acquis déjà une certaine hauteur quand il passe l'extrémité basse de la surface bitumée.
On estime que cette hauteur, pour être acceptable sans émoi excessif, doit être d'environ 1m. Des discussions avec des personnes ayant l'expérience de ces altisurfaces il résulte, qu'en hiver, une congère de 0, 50 à 1 m pourra subsister en bout de piste après déneigement c'est évidemment au-dessus de cet obstacle qu'il faudra ajouter la marge de 1 m mentionnée ci dessus.

Des figure 15 et 15b sur lesquelles nous avons tracé la hauteur au seuil en fonction de la distance de roulement, tous W/V confondus, on peut déduire la marge au roulement à prendre en compte pour passer cet "obstacle" au seuil.

Marge Hauteur au seuil (m)	Marge de roulement (m)		Marge de roulement % de Piste	
	MEGEVE	COURCHEVEL	MEGEVE	COURCHEVEL
0,5	26	35	6,5	9,5
1	46	35	11,5	9,5
1,5	50	40	12,5	11
2	63	60	15	16

Pour le DHC6 on a donc sensiblement les équivalences suivantes
Marge au seuil 1m si 10 % de marge au roulement
Marge au seuil 2m si 15 % de marge au roulement
Le calcul de la masse maximale au décollage en tenant compte de ces marges doit permettre d'éviter une rotation prematurée.

9.4 - Trajectoires au décollage - recherche d'un critère de performance

Une fois passé le seuil de piste il est important de continuer à s'intéresser à la trajectoire d'envol. Il est rapidement apparu, compte tenu de l'environnement dégagé, aux abords immédiats des altiports, que la trajectoire intéressante allait jusqu'au point où le taux de montée s'annulait. Les figures 16 et 17 donnent les trajectoires les plus bases constatées sur les pistes de COURCHEVEL et MEGEVE (ALPE D'HUEZ étant un altiport équivalent à COURCHEVEL) étant précisé que les W/V pratiqués allaient de 4,6 % à - 1 %).

On peut faire les remarques suivantes :

1 - W/V voisin de 2 % - Le pilotage ne pose pas de problème. L'accélération de VR vers V2 est franche, car on a naturellement tendance à continuer à descendre légèrement après décollage ; on ne rétablit un taux de montée positif qu'une fois V2 établie.

2 - W/V voisin de 0 % - L'accélération de VR à V2 est évidemment plus lente et la phase la plus critique se situe juste après le décollage lorsque l'avion survole la plateforme prolongeant la piste. Le pilote suit en gros la pente afin d'accélérer tout en conservant une certaine hauteur de 1 à 2 mètres par rapport au terrain. V2 atteinte, il se stabilise en palier. Dans l'environnement très dégagé de l'Alpe d'Huez ou de Courchevel la manoeuvre est acceptable. Sur le plan réglementaire la manoeuvre est acceptable, semble-t'il, à condition de passer le seuil à une certaine hauteur ($\simeq$ 1 m) et à Vi $\simeq$ V2.

3 - En prenant les marges au roulement recommandées, on constate que l'on obtient VZ = 0 toujours au dessus du seuil.

4 - Pour les W/V négatifs le pilote tend à établir instinctivement une trajectoire montante. Le taux de descente s'annule mais forcement au détriment de la tenue de la vitesse qui diminue. Ceci est dangereux et donc inacceptable en monomoteur réel.

Compte tenu de ces remarques il semble acceptable de conserver la limitation poids-pente au 1er segment donnant un W/V = 0 . Pour éviter aux autorités des altiports concernés a avoir à contrôler les obstacles sur une trajectoire descendante on peut suggérer que la trajectoire nette soit au dessus de celle présentée figure 18. Cette trajectoire nette se termine par une pente à 0 % - qui imposerait, selon les errements en vigueur pour les bimoteurs une pente brute de 0,8 % au deuxième segment. L'expérimentation effectuée confirme que ce W/V de 0,8 % reste acceptable dans l'environnement des altiports pratiqués.

9.5 - Conclusions et recommandations

Limitations
- Volets décollage 10°
- Vent arrière < 10 Kt
- Vent travers < 10 Kt

Masse maximale au décollage
- Limitations poids-pente inchangée au 1er segment
  0,8 % au 2eme segment
- Llof (panne moteur à VEF) < 0,9 Lrunway | 0,85 Lrunway | selon déneigement du seuil de piste

Procédure
- Décision au point de cassure de pente
- VR = 1,1 Vs (ou 1,1 VMCA) - 3Kt
- V2 = 1,2 Vs ou 1,1 VMCA

Tenir l'axe avec la dirigeabilité profondeur maintenue à piquer.

Equipage 2 pilotes qualifiés

9.6 - Type d'altiport recommandé

Au cours de ces essais effectués sur les 3 altiports de Megève, Courchevel, Alpe d'Huez avec plusieurs pilotes, il est apparu clairement que l'altiport d'Alpe d'Huez, était le plus satisfaisant. Sa cassure est assez loin, ce qui permet des vitesses de décision plus élevées et il n'est pas, contrairement à Megève et Courchevel, limité Poids - piste, ce qui permettrait une exploitation plus rentable.
Avec une zone latérale de dégagement en cas d'accélération arrêt, ce type d'altiport serait proche de l'optimum.

10) ATTERRISSAGE SUR ALTIPORTS

Nous avons profité de cette expérimentation pour effectuer également des mesures à l'atterrissage. Compte tenu de l'échantillon de pilotes ayant participé, les résultats statistiques obtenus peuvent être estimés représentatifs d'une population de pilotes qualifiés sur la machine et qualifiés montagne. A Megève et à Courchevel, les approches ont été effectuées alternativement avec et sans VASI. Les trajectoires ont été filmés au cinéthéodolithe (A l'Alpe d'Huez, il n'y a pas de VASI et il n'y avait pas le cinéthéodolithe). La procédure était toujours la même : Approche à
VREF = 67 kt avec les volets à 37°5.

10.1 - Précision d'impact

Distance Seuil - Impact (mètres)	MEGEVE		COURCHEVEL	
	Sans VASI	Avec VASI	Sans VASI	Avec VASI
Moyenne (mi)	92	64	104	114
Ecart Type σ i	20	20	45	21
mi + 2 σ i	52 - 132	24 - 104	14 - 194	72 - 156

Ces résultats montrent que l'atterrissage sans VASI est plus délicat à COURCHEVEL (plus dispersé). Un critère possible en approche manuelle est de prendre m + $2\sigma_i$ comme dispersion maximale en utilisation normale. Avec ce critère on voit qu'il reste encore de la marge par rapport à la piste, mais c'est juste à COURCHEVEL sans VASI. Ce critère qui est assez facile d'application (observateurs visuels) pourrait être retenu pour d'autres avions postulants à une utilisation TPP sur altiports.

10.2 - Distance de roulement à l'atterrissage

Il est assez irréaliste sur un altiport de mesurer la performance d'atterrissage comme étant celle obtenue en freinant au maximum. Un atterrissage normal se fait sans freiner, sauf sur la plateforme pour manoeuvrer, et il faut souvent remettre des gaz pour rouler et remonter la pente jusqu'à la plateforme.
Nous avons donc défini une distance de roulement comme étant la distance séparant l'impact du point où l'avion est contrôlé (environ à 30 kt), l'équipage effectuant les actions habituelles à l'atterrissage sur altiports notamment en passant les manettes de puissance sur ralenti reverse, sans jamais freiner ni appliquer la puissance en reverse (sauf nécesité, qui ne s'est pas produite).
Les résultats sont les suivants :

L roulement (m)	MEGEVE	COURCHEVEL	ALPE D'HUEZ
Moyenne mr	197	139	162
Ecart type σ r	30	18	19

10.3 - Longueur de piste nécessaire

A notre avis, il n'est pas utile de prendre en compte le σ_r du roulement dans l'analyse de la longueur de piste nécessaire à l'atterrissage. En effet, le pilote pourrait jouer sur σ en mettant plus ou moins de reverse ou de freins ·· il y a corrélation évidente entre la distance de roulement et le point d'impact. Il est par contre important de vérifier que la valeur moyenne du roulement telle que nous l'avo~s définie et mesurée permet d'arriver tranquillement (30 kt) au maximum) sur la platefor:.e.

La valeur de référence serait alors :

$mi + 2\,\sigma\,i + mr$

ce qui donne à Megève sans VASI 319 m (pour 403 m total)
à Courchevel sans VASI 333 m (pour 370 m total)

10.4 - Conclusion sur les atterrissages

Le DHC6 étant un avion très satisfaisant à l'atterrissage sur altiports, il ne faut pas perdre de vue que des avions moins "performants" pourraient encore être acceptables. Les critères proposés ci-dessus, ne le sont qu'à titre indicatif mais rendent compte néanmoins du fait qu'une marge supplèmentaire reste encore disponible.

Nous ne saurions trop insister sur l'importance du VASI qui a pour principal intérêt de permettre de stabiliser la pente d'approche et d'éviter les erreurs d'appréciation que la répartition à $m + 2\,\sigma_i$ proposée ci-dessus ne prend pas en compte. Il serait déraisonnable d'envisager une utilisation commerciale sans VASI (en particulier par conditions Météo moins favorables que celles volontairement recherchées pour cette expérimentation).

11) EXPLOITATIONS

Les autorisations d'exploitation ont été données cas par cas sur les bases de recommendations précédentes - en utilisant le modèle de calcul validé par les essais et donc applicable pour le moment au seul DHC6-300 (cf figure 19). Elles ne sont valables que sur des pistes séches en dur. Le problème de la marge du roulement a été résolu de façon conservative en considérant que la piste disponible utilisée pour les calculs au décollage = 0, 85 L runway.
Cette marge de 15 % est > aux recommandations précédentes. Pour l'exemple choisi
m = 5670 Kg Ts = 25° Z = 0 . On trouve LASD = Llof = 510 m avec V1 = 64, 3 Kt (piste balancée). Lrunway exigé = 1,15 x 510 = 586 m
Nous avons surtout parlé de décollage. Hors il faut bien également atterrir !
L'arrêté du 25 Novembre 1974 précise que sur les terrains présentant un dégagement suffisant la distance d'atterrissage doit être inférieure ou égale à 70 % de la longueur utile à l'atterrissage (Lrunway). La figure 20 montre que cette exigence est compatible avec les limitations au décollage ce que re... des pistes de 600 m exploitables par un DHC6-300.

Il faut insister sur le fait que pour le moment ces autorisations d'exploitations sont données cas par cas après examen par les autorités opérationnelles de la DGAC des particularités propres à chaque terrain et à chaque compagnie concernés.

L'examen des problèmes qu'aurait pu posé ces procédures est prévu après une période d'exploitation de 6 à 12 mois c'est à dire dans les mois qui viennent. Une réduction de la marge de 15 % à 10 % sera alors examinée.

12) CONCLUSION

Les études menées par la DGAC avec l'aide de DHC et des utilisateurs ont permis de définir des procédures qui garantissent à priori un niveau de sécurité satisfaisant en ce qui concerne la panne de moteur au décollage sur des pistes courtes utilisées par des avions multimoteur de TPP2 (masse inférieure à 5700 Kg aménagés pour 10 pax et plus)
Les autorisations d'exploitation sur ces pistes selon ces procédures étant très récentes il est prématuré de tirer des conclusions définitives sur les résultats opérationnels
Les dérogations accordées ne concernent pour le moment que l'avion DHC6-300.
Les recommendations formulées seraient cependant applicables à d'autres types d'avions à condition de disposer des données complémentaires concernant la vitesse minimale de contrôle au sol et les performances de roulement au décollage avec un moteur en panne.

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2 - M. COLLOT — Expérimentation atterrissage de nuit sur altiports
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CEV/Is/SE/AV 1772 : 1980

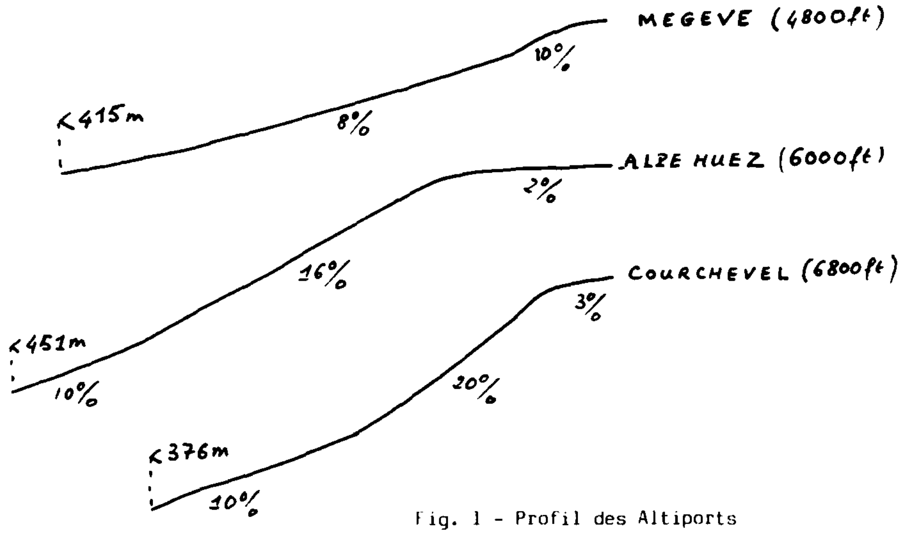
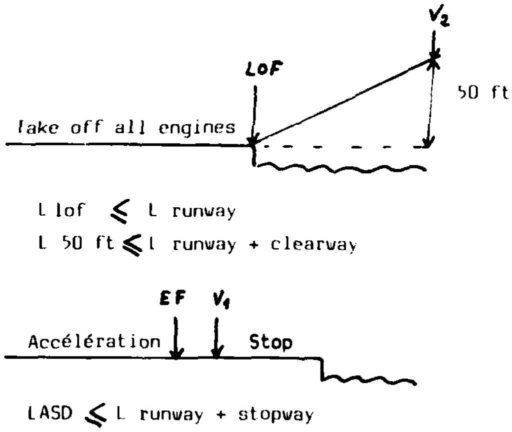
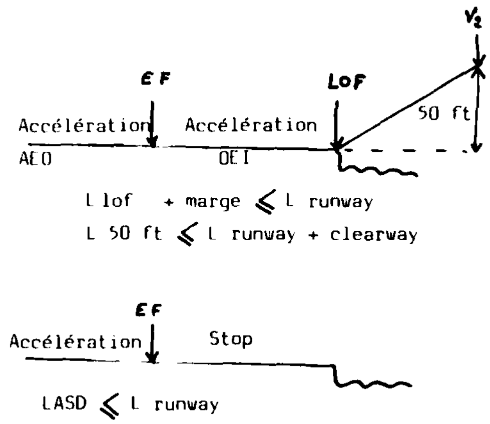


Fig. 1 - Profil des Altiports



V1/VLOF	⩾	1,1 VMCA/1,1 VS
V2	⩾	1,1 VMCA/1,2 VS

Fig 2 - Règlement Applicable



VEF	VMCG
VLOF	1,1 VMCA/1,1 VS
V2	1,1 VMCA/1,2 VS

Fig. 3 - Dérogation possible

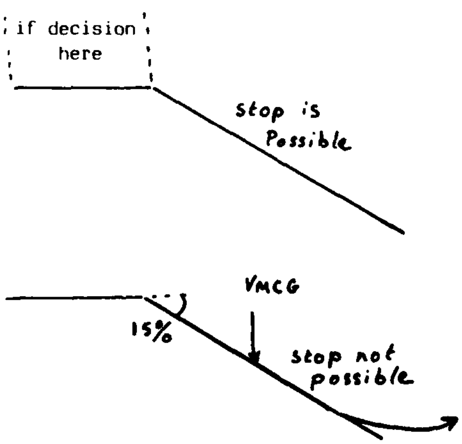
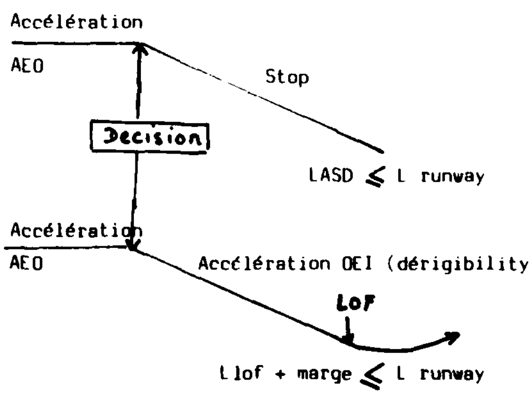


Fig. 4 - Cas particulier Altiport



$\boxed{\text{VLOF} \geqslant 1{,}1\ \text{VMCA}/1{,}1\ \text{VS}}$

Fig. 5 - Dérogation possible Altiport

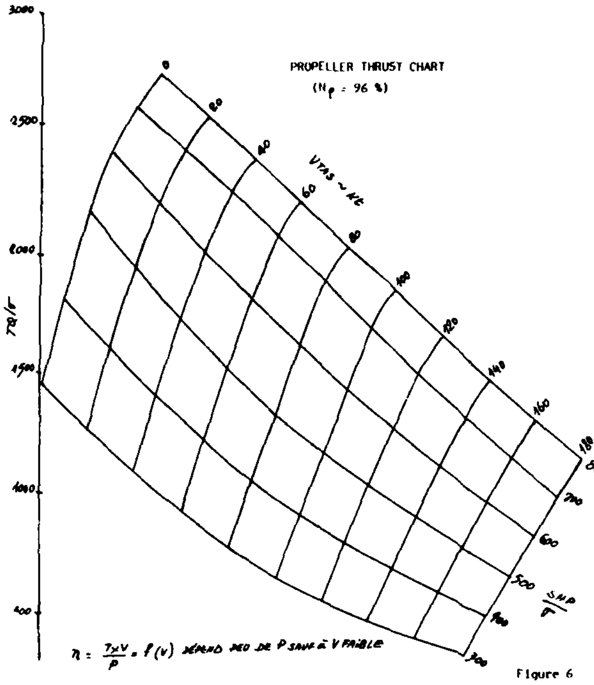


Figure 6

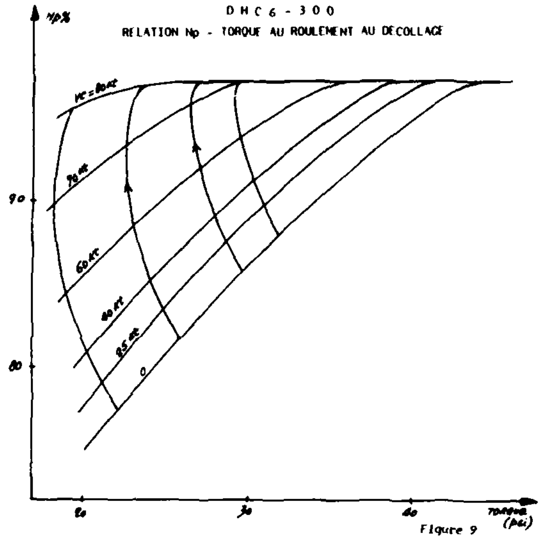


Figure 9

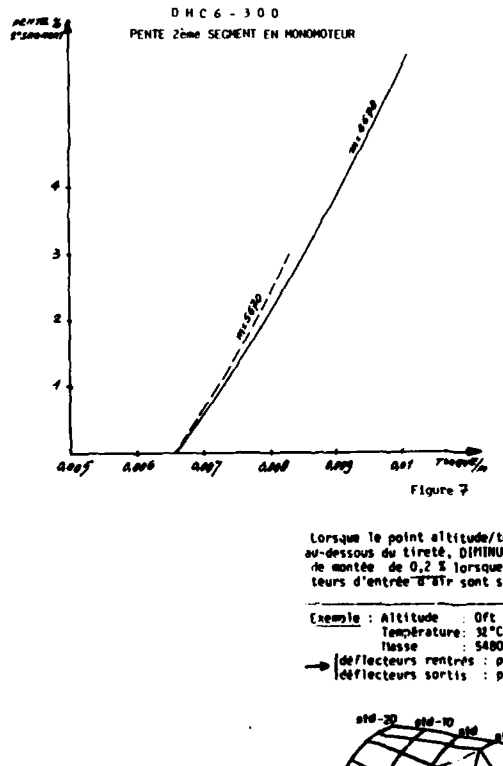


Figure 7

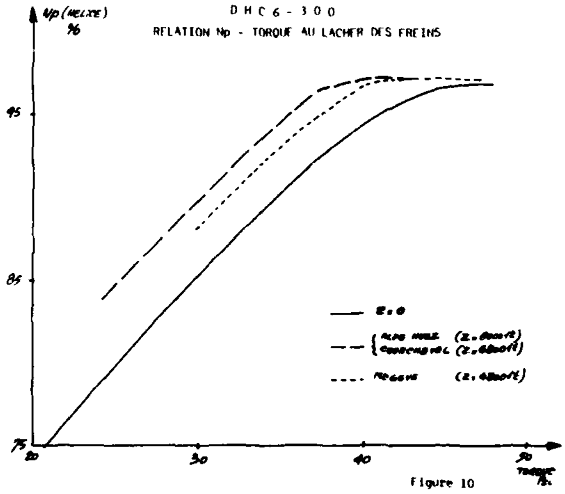


Figure 10

PENTE DE MONTEE - 1 MOTEUR PUISSANCE DECOLLAGE

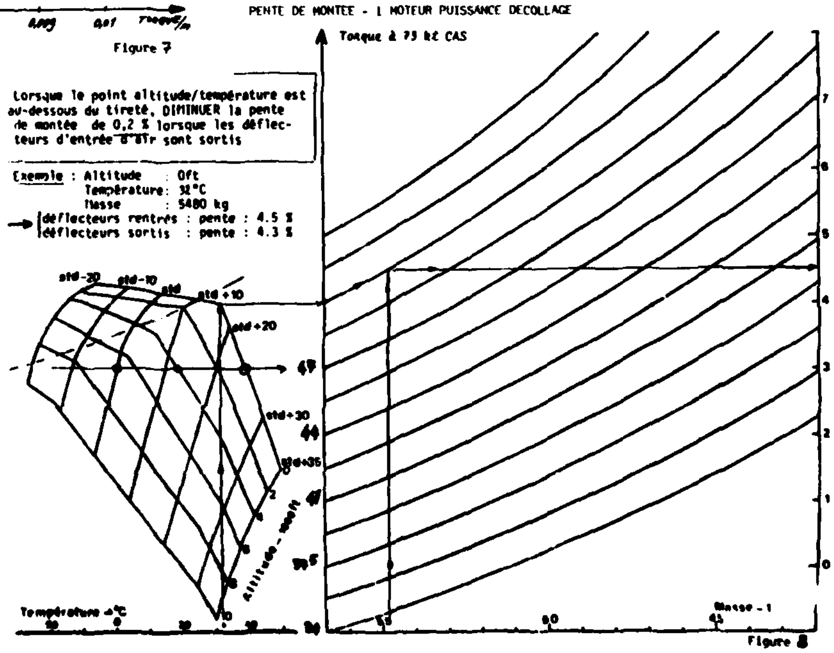


Figure 8

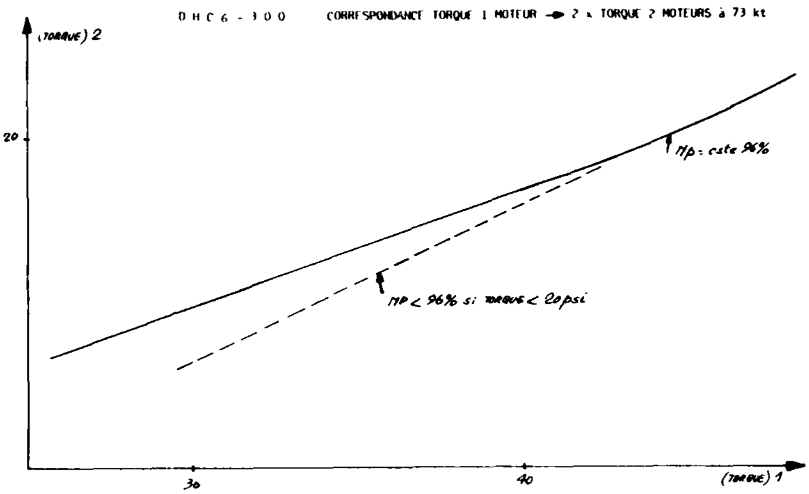


Figure 11

DHC6-300

VITESSE MINIMALE DE CONTROLE SOL (Gouvernes aérodynamiques seules) Volets 10° - Volets 20°

Puissance au lâcher des freins N = 96% Torque = 48,5 psi

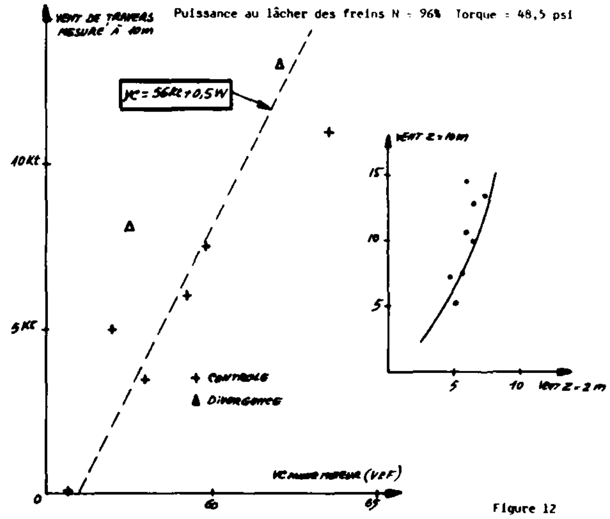


Figure 12

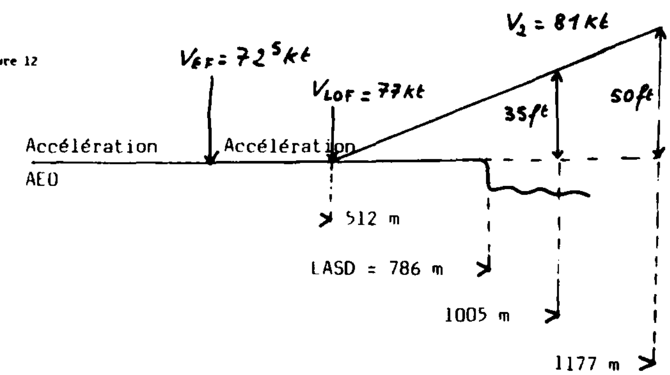


Fig. 13 - Take off distance OEI

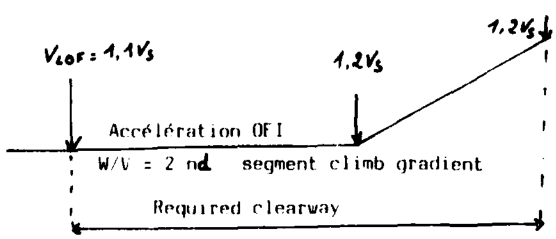


Fig. 14 - Clearway approximation

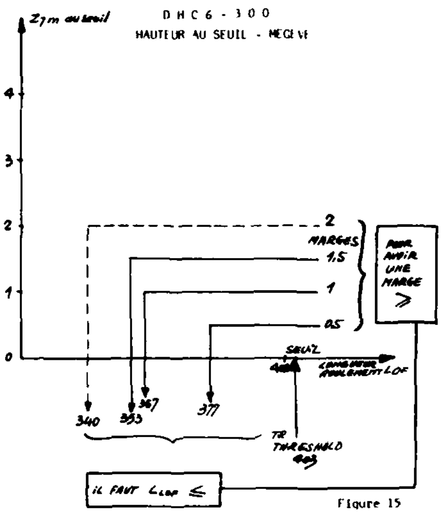


Figure 15

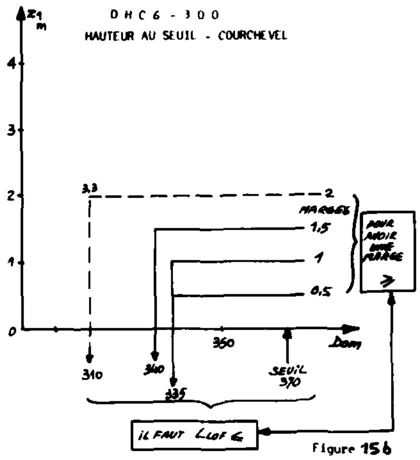


Figure 15b

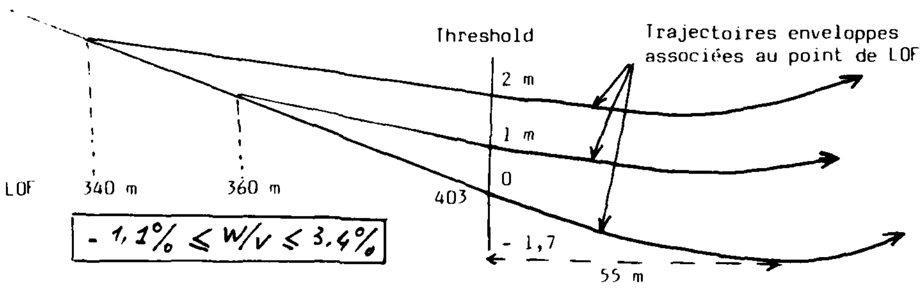


Fig. 16 - Trajectoires de décollage MEGEVE

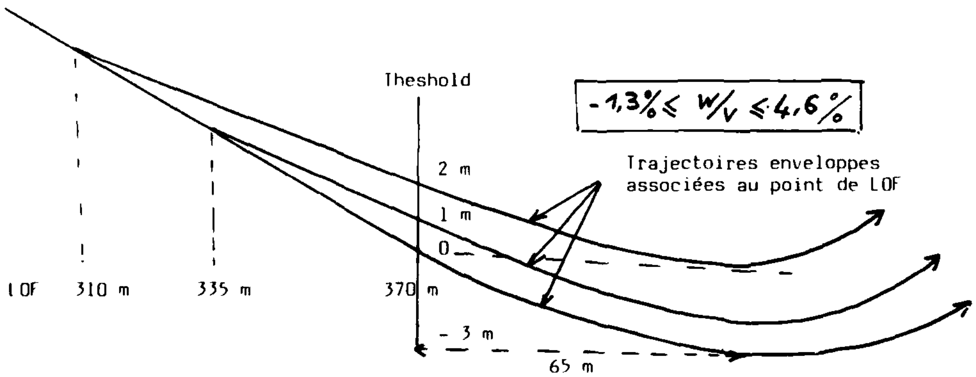


Fig. 17 - Trajectoires de décollage COURCHEVEL

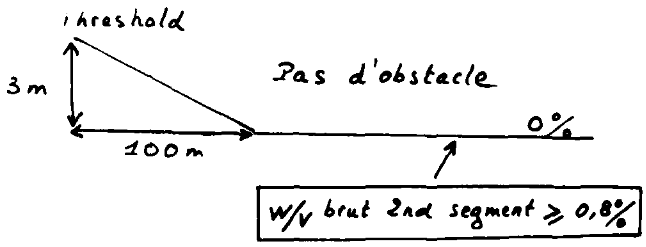


Fig 18 - Plan net d'obstacle : Altiport

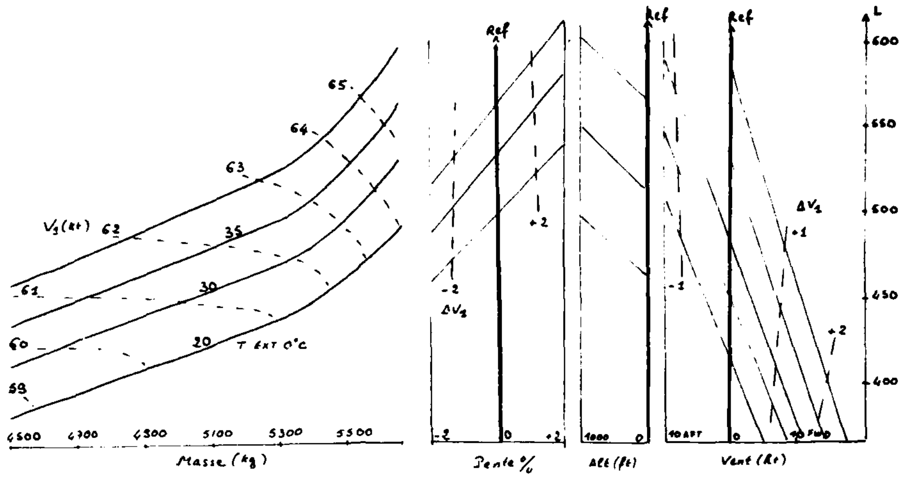


Figure 19   D H C 6 - Performance Chart.

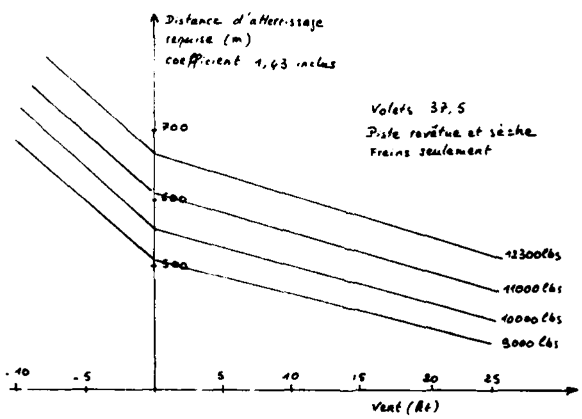


Figure 20   D H C 6 - 300   Longueur de piste nécessaire à l'atterrissage.

CARACTERISATION SYSTEMATIQUE DES EFFETS DE L'ELECTRICITE ATMOSPHERIQUE SUR LES CONDITIONS OPERATIONNELLES DES AERONEFS

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RESUME

L'introduction des commandes de vol électrique et la généralisation des matériaux structurels en composite ont rendu nécessaire une évaluation précise de la menace de l'électricité atmosphérique sur les conditions opérationnelles des aéronefs. La caractérisation systématique de cette menace exige la réalisation d'essais en vol utilisant des avions à l'instrumentation particulièrement élaborée. Les résultats obtenus ne peuvent être exploités avec profit que grâce à une confrontation permanente des équipes à l'échelle internationale organisée par le biais de Conférences spécialisées, ou, mieux, d'accords bilatéraux comportant des échanges de données et de personnel.

ABSTRACT

The introduction of fly-by-wire and Active Control Technology and the extensive use of composite materials have emphasized the need for a precise assessment of atmospheric electricity impact on aircraft operation, or, in other words, of the threat due to static charging and lightning on the avionic systems. To characterize this threat in a systematic way, flight tests have to be performed using aircraft instrumented with very sophisticated measurement systems. For both static electricity and lightning, the philosophy of this type of test is discussed, starting from the physical mechanisms involved, analyzing the nature of the threat, mentioning the hardening methods and, more generally, giving the backbone of the research and development effort needed to solve the problem. The scope of systematic characterization by flight tests with specially instrumented aircraft is defined, and the conditions of such operations are given in detail. A general conclusion emerges from this analysis : to increase the amount and the relevance of the data from which a world-wide model is extracted, it is necessary to compare measurements, to criticize methods and to fit the results into an unique set : this requires an international cooperation by the channel of specialist meetings or bilateral agreements involving exchange of data and personnel.

I - GENERALITES

Trois facteurs ont attiré l'attention, dans la dernière décennie, sur les interactions électricité atmosphérique/aéronef :

a) la nécessité de permettre le vol quelles que soient les conditions météorologiques ;
b) la généralisation des matériaux composites dans la structure ;
c) la mise en oeuvre des commandes de vol électriques et, plus généralement, le développement des techniques de Contrôle Automatique Généralisé (CAG).

Ces facteurs augmentent la vulnérabilité des aéronefs aux effets de l'électricité atmosphérique (électricité statique et foudre).

Le souci de réduire cette vulnérabilité a suscité, depuis une dizaine d'années, un important effort de recherche et de développement. Dans le cadre de cet effort, il est nécessaire de caractériser de façon systématique la menace que constitue l'électricité atmosphérique pour le vol des aéronefs. En particulier, si l'on considère le problème de la sensibilité de l'Avionique aux perturbations électromagnétiques, il est intéressant de comparer la menace due à la foudre et celle due à l'électricité statique. Cette caractérisation est indispensable si l'on veut dimensionner correctement, aussi bien en poids qu'en coût, les dispositifs de protection chargés d'assurer la survie du système dans les conditions les plus sévères que l'aéronef est susceptible de rencontrer.

Pour effectuer cette caractérisation, l'utilisation d'essais en vol sur des appareils spécialement instrumentés s'est avérée nécessaire. Le but de la présente synthèse est de définir l'objectif précis et les limites de ce type d'essais en vol dans l'ensemble du processus de recherche et développement devant aboutir à la définition d'une protection équilibrée des aéronefs contre les dangers de l'électricité statique, de la foudre, et, éventuellement, de l'impulsion électromagnétique.

II - INTERACTION ELECTRICITE STATIQUE/AERONEF

Nous étudierons d'abord le problème posé par l'accumulation sur la cellule, dans certaines conditions de vol, d'électricité statique. Pour cela, nous rappellerons succinctement les mécanismes physiques responsables du phénomène, définirons la nature de la menace pour les aéronefs, citerons les techniques de protection d'usage courant,

ainsi que la méthodologie employée pour améliorer cette protection ; nous situerons enfin, sur un exemple précis, l'intérêt des méthodes d'essais en vol en soulignant les résultats récemment obtenus au cours de tels essais.

II.1 - Mécanismes physiques de charge des aéronefs

Nous retiendrons essentiellement deux mécanismes :

a) lorsque l'aéronef rencontre des précipitations solides, le contact des aérosols initialement neutres avec la cellule leur confère une charge électrique, la charge opposée étant laissée sur l'aéronef ; c'est ainsi que les aérosols de glace chargent la cellule négativement, le courant correspondant pouvant atteindre plusieurs milliampères pour un avion gros porteur ; les précipitations liquides donnent, par suite de l'éclatement des gouttes d'eau à la suite de l'impact, une charge positive à l'aéronef ;
b) lorsque l'aéronef se trouve situé au voisinage de nuages électriquement chargés, il se crée par influence électrostatique une séparation des charges libres à la surface de la cellule ; le champ électrique superficiel peut atteindre, en certains points, une valeur suffisante pour amorçer une décharge corona ; en général, des décharges des deux signes sont amorcées, mais le système n'est pas suffisamment symétrique pour que les courants positifs et négatifs de ces décharges se compensent : la différence contribue donc à charger électriquement la cellule.

Pour plus de détails sur les mécanismes de charge des aéronefs, nous renvoyons à la littérature [1] [2] [3].

II.2 - Nature de la menace due à l'électricité statique

L'énergie électrostatique accumulée sur la cellule a une valeur très faible et ne serait absolument pas gênante si elle ne se transformait pas en énergie électromagnétique par amorçage de décharges électriques à partir d'une valeur critique du champ électrique produit par l'accumulation de ces charges statiques. Cette énergie électromagnétique est rayonnée sous forme de parasites radioélectriques sur les antennes des sytèmes de radio-communications et de radio-navigation de l'aéronef, et peut gravement perturber le fonctionnement de ces systèmes critiques pour la sécurité du vol.

L'exposition à des conditions météorologiques de nature à provoquer un courant de charge électrique peut être de longue durée ; la phase de mauvais fonctionnement des systèmes incriminés survient toujours lorsque ces systèmes seraient le plus utiles (vol aux instruments) : il s'agit donc d'une menace relativement peu brutale mais particulièrement insidieuse. Ce danger se manifeste surtout pour les aéronefs de petite dimension, avions civils d'affaires ou de tourisme, appareils militaires d'entraînement, hélicoptères, dont les antennes sont relativement proches des sources de parasites.

Trois types de décharges sont provoqués par l'accumulation d'électricité statique (fig. 1) [4] :

a) les étincelles, entre conducteurs métalliques portés à des potentiels électriques différents par suite d'effets de charge différentielle ;
b) les décharges glissantes, sur les surfaces isolantes chargées (fig. 2) ;
c) les décharges corona, au niveau des pointes et des angles vifs conducteurs de la structure.

Ces décharges ont un courant impulsif, donc susceptible de rayonner une forte énergie électromagnétique ; à titre d'exemple, la décharge corona négative se caractérise souvent par un régime de relaxation formé d'impulsions dites de Trichel, dont la forme est donnée sur la figure 3 (double exponentielle de temps de montée 10 ns et de constante de temps de descente 100 ns) ; le spectre correspondant est donné par la figure 4 ; on voit que l'énergie rayonnée est importante dans le domaine de fonctionnement de la navigation Oméga (VLF), du radio-compas (HF) ; elle est moins importante dans le domaine du VOR et de l'ILS (VHF) et devient négligeable pour le MLS (UHF et micro-ondes). C'est pourquoi les appareils de l'aviation générale, les moins bien équipés pour des raisons économiques en moyens de radio-navigation à fréquence élevée, sont aussi les plus vulnérables à la menace de l'électricité statique. En particulier, les radio-compas sont paralysés par saturation de leurs boucles de réaction sous l'effet de décharges glissantes qui s'établissent par suite de l'accumulation de charges statiques sur les carènes d'antenne. Le même phénomène peut provoquer des erreurs de fonctionnement des circuits associés au VOR-ILS de certains avions d'entraînement [5] [6] [7].

II.3 - Techniques de protection d'usage courant

Dans l'ensemble, les techniques de protection à appliquer sont connues depuis de nombreuses années :

a) pour supprimer les étincelles, il suffit de pratiquer la métallisation de l'appareil, c'est-à-dire relier électriquement toutes les parties conductrices ;
b) pour supprimer les décharges de surface, il faut recouvrir de revêtements antistatiques les parties isolantes susceptibles d'être chargées par le contact des précipitations ; ces revêtements sont des couches résistives assez conductrices pour empêcher l'accumulation des charges statiques mais assez résistantes pour ne pas compromettre la transparence radio-électrique des radômes et carènes d'antenne ;

c) les décharges corona ne peuvent pas être supprimées, car elles participent à l'équilibre électrique de l'aéronef ; on se contente donc de diminuer leur couplage avec les antennes ; pour cela, on les éloigne de la masse de la cellule en favorisant leur amorçage à l'extrémité de pointes métalliques fines reliées à la structure par une résistance de valeur élevée (découplage résistif) ; on peut aussi orienter la ligne de courant de ces décharges perpendiculairement à la ligne de force du champ électrique que rayonnerait l'antenne fonctionnant comme émettrice (découplage orthogonal) ; on peut enfin remplacer la pointe métallique par l'extrémité d'un faisceau de fibres de carbone ; on montre que, dans ces conditions, chaque impulsion de Trichel est remplacée par un grand nombre de micro-impulsions décorrélées, ce qui diminue le bruit radio-électrique à la source ; les dispositifs correspondants sont appelés *déperditeurs passifs* (fig. 5) [4] [8].

L'utilisation de déperditeurs passifs peut se heurter à des difficultés particulières dans le cas des hélicoptères en vol stationnaire ; par ailleurs, il peut être difficile d'éviter les décharges corona localisées à l'extrémité des pales d'une hélice : nous ne traiterons pas ici complètement le cas des avions à hélice et des hélicoptères ; certaines de nos conclusions ne s'appliquent en toute rigueur qu'au cas des avions munis de réacteurs.

II.4 - Méthodologie d'optimisation de la protection électrostatique

La définition et la mise en oeuvre des dispositions destinées à assurer la protection des aéronefs contre les effets des charges électrostatiques ne sont pas des opérations triviales : le succès dépend de l'application rigoureuse d'une méthodologie dont les éléments essentiels ont été formulé par Nanevicz et al, et qui a été récemment complétée par les travaux de l'ONERA [[9] [10].

La première étape de la protection consiste à effectuer une métallisation convenable ; pour orienter ce travail, il est nécessaire de pouvoir mesurer la résistance électrique entre parties métalliques de l'aéronef même si ces parties sont recouvertes extérieurement de peinture isolante ; c'est le rôle d'un appareillage spécialisé fonctionnant par capacité à travers la couche de peinture, le CORAS. L'intervention de cet instrument permet de repérer très rapidement des défauts de métallisation indécelables par les moyens classiques.

La deuxième étape consiste à recouvrir les parties isolantes susceptibles de subir l'impact des précipitations par un revêtement antistatique. Là encore, le CORAS est utilisé, aussi bien au cours de l'élaboration à l'atelier des couches antistatiques que pour des vérifications sur le terrain ; il permet en effet de connaître la valeur de la résistance de surface de ces couches, même si elles sont protégées par une couche isolante de peinture de finition.

L'efficacité de ces mesures de protection contre les étincelles et les décharges de surface est ensuite testée au sol par simulation ; on peut utiliser pour cela un injecteur de charges électrostatiques, l'INJECO, qui projette localement, sous forme d'aérosol de glace chargé, un courant pouvant aller jusqu'à 100 µA. Cette méthode locale est efficace pour déceler les anomalies provenant d'une protection insuffisante contre la charge différentielle entre les différentes parties de la cellule, qui constitue la cause principale de perturbation.

Il faut ensuite passer à la mise en place des déperditeurs passifs, destinés à limiter les effets des décharges corona sur les antennes. Cette mise en place est prévue de façon qu'aucune décharge corona ne s'amorce sur des parties non protégées de la cellule ; elle doit tenir compte des facteurs suivants : seuil d'amorçage des décharges sur les déperditeurs, courant maximum du déperditeur pour définir leur nombre en fonction du courant de charge maximum escompté, influence réciproque entre déperditeurs définissant une distance minimum entre ces dispositifs. Bien entendu, la connaissance du champ électrostatique à la surface de la cellule est indispensable pour effectuer cette étude. La mise en place des déperditeurs est enfin affinée en tenant compte des effets de résonance dimensionnelle de l'aéronef dans la gamme des fréquences de travail des antennes : il faut en effet éviter des effets d'amplification du bruit radioélectrique des décharges corona par ces résonances.

Lorsque l'implantation des déperditeurs a ainsi été effectuée, il est utile d'effectuer un test pour évaluer son efficacité. On procède alors à un essai au sol, en isolant l'aéronef et en le portant à un potentiel électrique élevé. L'expérience montre que ce test est difficile à réaliser, car il est pratiquement impossible de reproduire au sol le champ électrique existant en vol. Par ailleurs, l'écoulement aérodynamique introduit en vol des variations de pression qui modifient localement les conditions de décharges corona. On est donc conduit à effectuer un essai en vol.

II.5 - Caractérisation systématique en vol de l'impact de l'électricité statique sur le fonctionnement des systèmes de radio-communication et de radio-navigation d'un aéronef

Nous choisirons l'exemple d'une campagne d'essais en vol réalisée en France sur un avion METEOR spécialement instrumenté (fig. 6). L'annexe I donne les conditions précises de cette campagne. Nous soulignerons simplement ici les faits saillants qui

permettent de situer le caractère général et de dégager les apports essentiels de ce type d'essai.

1) L'aéronef doit avoir subi le traitement complet de métallisation et de traitement antistatique, pour éliminer tout effet d'étincelle et de décharge de surface.

2) L'aéronef doit être très fortement instrumenté [11]. Parmi les paramètres qu'il est nécessaire de mesurer, il faut citer :
- le courant d'impact tribo-électrique, mesuré par des sondes d'impact, petites surfaces collectrices affleurantes localisées sur la zone frontale de l'aéronef ;
- le courant traversant les déperditeurs passifs ;
- le potentiel (ou la charge électrique totale) de l'aéronef ;
- le champ électrique atmosphérique (trois composantes) ;
- le bruit radio-électrique induit sur les antennes.

L'évaluation du potentiel et celle du champ extérieur sont effectuées par analyse des mesures de champ électrique superficiel effectuées grâce à une disposition convenable de cinq moulins à champ montés affleurants sur la cellule (on utilise cinq moulins pour avoir de la redondance).

3) L'aéronef doit voler dans des conditions représentatives des conditions atmosphériques les plus défavorables pour le vol aux instruments, en tentant de trouver des circonstances où les différents facteurs sont bien séparés ; le pilote cherchera, en particulier, à caractériser séparément l'effet de l'impact et celui de l'influence électrostatique des nuages d'orage.

4) On cherche à comparer les résultats obtenus sur l'aéronef muni de sa protection optimale en déperditeurs avec les résultats obtenus sur le même aéronef sans cette protection.

5) Dans ces conditions, les essais font apparaître les possibilités et les limitations de l'état de l'art en matière de protection des aéronefs contre l'électricité statique.

L'analyse de l'ensemble des vols effectués en présence d'impact, mais sans champ électrique externe, fait apparaître que le bruit radio-électrique, qui atteint dans la gamme de fonctionnement du radio-compas des valeurs de champ électrique de l'ordre de 3 mV/m sans déperditeur, est ramené par l'action des déperditeurs au-dessous de 80 µV/m. L'efficacité des protections contre l'effet de l'électricité statique est donc démontrée dans ce cas. La situation est cependant modifiée en présence de champ électrique atmosphérique.

Sur la figure 7, on remarque une phase initiale caractérisée par un champ extérieur et une montée du potentiel en absence d'impact ; en même temps, le bruit radio-électrique est notable ; il ne semble pas que dans ce cas, les déperditeurs aient joué leur rôle. L'interprétation du phénomène est la suivante : le champ électrique extérieur est suffisamment fort et son orientation par rapport à l'aéronef suffisamment éloignée des plans de symétrie pour que les points de champ électrique superficiel maximal ne soient plus à l'extrémité des déperditeurs : dans ces conditions, des décharges corona incontrôlées s'amorcent directement sur la structure.

Une telle configuration aurait pu passer totalement inaperçue sans essai de caractérisation systématique en vol. Les conclusions de l'étude, en attendant la mise en oeuvre d'un moyen opérationnel destiné à réduire la probabilité d'apparition de décharges parasites directement sur la structure, est que le pilote doit chercher à éviter la proximité immédiate de certains types de nuages caractéristiques de l'orage (cumulus congestus) lorsqu'il utilise ses aides à la navigation ; par contre, dans un aéronef convenablement traité, il peut les utiliser en toute quiétude au milieu des précipitations en absence de champ électrique extérieur. On en déduit aussi que, dans un aéronef convenablement traité, le bruit radio-électrique résiduel qui se traduirait, dans les écouteurs, par un bruit de friture, pourrait être utilisé par le pilote comme indice de proximité d'un système nuageux susceptible de provoquer précipitation de grêle ou foudroiement.

III - INTERACTION FOUDRE/AERONEF

Les mécanismes physiques essentiels du foudroiement d'un aéronef seront rappelés succinctement dans ce chapitre. La nature de la menace sera définie, en insistant particulièrement sur la menace vis-à-vis de l'Avionique. Après avoir décrit le principe des mesures de durcissement, nous indiquerons les grandes lignes des travaux de recherche et de développement qui doivent aboutir à une protection optimisée de l'aéronef contre la foudre.

Les travaux de caractérisation, qui ont pour but de fournir les renseignements de base pour la recherche technique, comportent une partie importante d'essais systématiques en vol. On indiquera leur objectif précis, les conditions de leur réussite, et l'on citera quelques uns des résultats préliminaires déjà obtenus.

III.1 - Mécanisme physique associé au foudroiement d'un aéronef

La foudre est le phénomène brutal de décharge électrique dans l'atmosphère par

lequel les charges électriques positives et négatives accumulées dans les nuages, ainsi que leurs images électriques au sol, se neutralisent. Les foudroiements correspondants peuvent être intra-nuages, inter-nuages, ou nuage-sol. Ces foudroiements sont précédés par une période de pré-claquage (pré-breakdown) à l'intérieur du nuage, puis par la progression d'un canal ionisé dit précurseur (leader) qui établit un pont conducteur entre centres de charges opposées. Dès que le circuit est fermé, un courant de court circuit intense le traverse à une vitesse qui est une fraction de celle de la lumière. C'est le coup en retour (return stroke), responsable des effets lumineux de l'éclair et qui provoque l'onde de choc perçue sous forme de tonnerre.

La phase de pré-claquage peut durer une fraction de seconde, la progession du leader quelques millisecondes, le coup en retour quelques dizaines de microsecondes ; un même éclair peut comporter plusieurss dizaines de précurseurs et coups en retour pendant une durée totale de l'ordre de la seconde ; dans de nombreux cas, le canal ionisé est parcouru entre les coups en retour par un courant persistant. Le précurseur peut transporter 500 Ampères, le coup en retour est inférieur à 200.000 Ampères dans 98 % des cas ; le courant persistant peut valoir plusieurs centaines d'Ampères.

Un aéronef peut être frappé par la foudre pour plusieurs raisons :

a) dans un premier scénario, un précurseur descendant passe à proximité de l'appareil ; comme il s'accompagne d'un champ électrique très intense, il amorce sur l'aéronef une décharge (streamer) qui se transforme en précurseur ascendant ; les deux précurseurs, qui comportent de nombreuses branches, se rencontrent ; l'appareil est porté au potentiel du nuage et émet vers le sol un précurseur descendant ; dès que ce précurseur est en contact avec le sol, le coup en retour remonte vers l'aéronef (fig. 8) ;
b) un second scénario est aussi possible : l'avion, plongé dans le champ électrique très intense d'un nuage d'orage, voit s'amorcer sur ses pointes une décharge qui fournit un précurseur ascendant vers le nuage : la suite des évènements se passe comme dans le premier scénario, mais ici le coup de foudre a été déclenché par la présence même de l'aéronef.

III.2 - Nature de la menace due à la foudre

Contrairement à la menace due à l'électricité statique, la menace de la foudre se caractérise par une action d'une extrême brutalité mais de faible probabilité. Les statistiques font état d'une fréquence moyenne de foudroiement d'une fois toutes les 3000 heures de vol pour les avions commerciaux. En général, sur la génération actuelle d'aéronef, les conséquences sont mineures ; cependant, l'USAF a recensé la perte de sept appareils au cours de la dernière décennie, et des dégats matériels d'un montant de 21 millions de dollars de 1972 à 1977 du fait de la foudre. En effet, bien que les méthodes de protection contre les effets du coup de foudre direct sur la structure aient été depuis longtemps mis au point, il subsiste un petit nombre de cas qui se soldent par des dégats sérieux et un nombre très réduit d'accidents catastrophiques (inflammation du combustible dans les réservoirs, éclatement du radôme, extinction du réacteur). Ces effets et les remèdes préconisés ont fait l'objet d'une abondante littérature spécialisée ; nous n'en parlerons pas ici.

La généralisation des matériaux composites, moins efficaces que les panneaux métalliques comme écrans électro-magnétiques, et l'introduction de commandes de vol électriques, avec leur avionique digitale sensible aux perturbations radio-électriques, transposent le problème sur un nouveau plan. La partie la plus vulnérable de l'aéronef devient alors cette avionique destinée à assurer des fonctions critiques pour la survie de l'appareil ; il convient donc de la protéger non seulement des conséquences du coup de foudre direct, mais aussi des rayonnements parasites en provenance des foudroiements de proximité, c'est-à-dire des effets électro-magnétiques des coups de foudre survenant dans un rayon de quelques dizaines de kilomètres. Ce problème, qui constitue un obstacle à franchir avant le développement du Contrôle Automatique Généralisé, a suscité l'effort technologique consacré à la foudre au cours de ces dernières années.

III.3 - Techniques de protection de l'avionique contre les effets du foudroiement

Nous ne ferons, dans cette synthèse, qu'effleurer ce problème en citant, parmi les techniques de protection (ou de durcissement), les différents moyens de réduire le couplage entre foudre et circuits sensibles : blindages électro-magnétiques, réduction de la surface des boucles collectrices de flux magnétique par utilisation de paires torsadées et par un choix judicieux des mises à la masse, utilisation des circuits opto-électroniques pour la transmission des informations sur fibre optique au lieu des fils conducteurs, etc.

Notons que l'optimisation de ce durcissement, c'est-à-dire la définition de dispositions efficaces au poids et au coût minimaux, requiert une connaissance quantitative de la source de perturbations, c'est-à-dire des niveaux et du spectre de la puissance rayonnée par les différentes phases du coup de foudre. En particulier, comme la menace est constituée par le champ parasite induit, la grandeur critique est liée à la vitesse de montée du courant de foudre dI/dt et non pas au courant I lui-même ou à la charge neutralisée Q comme c'est le cas lorsqu'on examine le problème des dégats mécaniques ou thermiques provoqués sur la cellule.

III.4 - Méthodologie de la recherche et du développement dans le domaine de la protection des aéronefs de la prochaine génération contre la foudre

La structure logique selon laquelle sont organisés la recherche et le développement dans le domaine qui nous intéresse est schématisée sur la figure 9. On relève les étapes suivantes : caractérisation des paramètres de la foudre, étude du couplage à l'aéronef, méthodes de durcissement, méthodes de test au sol et moyens d'essai correspondants, recommandations et normes. Dans cet ensemble, le but du présent exposé consiste à situer dans l'ensemble des travaux de caractérisation ceux qui utilisent des essais en vol systématiques sur aéronefs spécialement équipés. Nous renvoyons le lecteur intéressé par les autres aspects de ces travaux à la littérature spécialisée [12] [13] [14].

Notons cependant, en liaison avec la conclusion du paragraphe précédent, que les statistiques sur lesquelles les recommandations actuellement utilisées sont fondées proviennent principalement de mesures faites au sol dans des observatoires, il y a un certain nombre d'années. Ceci indique d'une part que l'on manque de données concernant les phénomènes perçus à l'altitude de vol des avions, et, d'autre part, que l'ensemble de ces travaux a été réalisé avec des instruments de mesure en grande partie périmés aujourd'hui, en particulier en ce qui concerne leur bande passante. C'est dire que si les renseignements qu'on en tire sont suffisants pour estimer la menace sur les structures, liées à I et Q, ils peuvent se révéler trompeurs pour l'évaluation des effets sur l'Avionique, liés à dI/dt, qui requiert une très large bande passante (50 ou même 200 MHz dans les chaînes de mesure utilisées à présent).

Il est donc nécessaire de faire en vol des mesures très précises, qui doivent permettre de définir les valeurs maximales de dI/dt, et plus particulièrement de confirmer les modèles établis par les théoriciens pour rendre compte de l'évolution des champs électriques et magnétiques de l'éclair, ou éventuellement de compléter ces modèles qui, à l'heure actuelle, ne concernent que le coup en retour [15]. Ces modèles sont en effet critiques pour le choix d'une protection équilibrée contre les menaces de la foudre, de l'électricité statique et, éventuellement, de l'impulsion électro-magnétique [16] ; il s'agit donc de mesures dont l'analyse entraînera des conséquences importantes sur la philosophie des méthodes de durcissement.

III.5 - Caractérisation systématique en vol des paramètres électro-magnétiques des foudroiements directs et de proximité

Il existe actuellement trois programmes en cours orientés vers cet objectif : un programme conduit par l'USAF (Wright Patterson Air Force Base) sur un C.130, un programme mené par la NASA (Langley Research Center) sur un F.106 B, et un programme développé en France sous l'égide du STTE sur un C.160 TRANSALL. Des essais systématiques même nature mais avec un appareillage réduit ont été conduits, il y a quelques années, par l'USAF sur un U2 et, plus récemment, par la NASA, en association avec le SRI sur un Learjet ; nous ne les discuterons pas dans cet exposé.

Les grandes lignes des trois programmes en cours apparaissent dans l'annexe II. Nous soulignerons simplement dans ce paragraphe les traits communs qui différencient des essais en vol classique ces opérations systématiques de caractérisation.

1) L'équipement de l'avion doit avoir été spécialement étudié au point de vue de la compatibilité électromagnétique ; c'est un véritable laboratoire volant, comprenant une enceinte étanche aux rayonnements dus à la foudre : il faut en effet que les chaînes électroniques délicates ne soient pas soumises à l'action parasite des perturbations qu'elles sont chargées de mesurer. La solution générale consiste à ménager un laboratoire parfaitement blindé à l'intérieur d'une cage de Faraday, où les informations pénètrent par fibre optique.

2) L'aéronef doit être équipé de nombreux capteurs. Parmi les paramètres qu'il est nécessaire de mesurer, citons :

- le courant de foudre (sur perche) ou les courants de peau induits sur la surface externe de l'appareil ; on mesurera ces courants ou leur dérivée par rapport au temps ;
- les champs magnétiques et électriques, ou leurs dérivées, en un certain nombre de points choisis à la surface extérieure de l'avion ;
- le champ électrique statique et le potentiel de l'avion (ces données sont utiles si l'on veut comprendre les mécanismes de déclenchement du foudroiement ; elles sont sans intérêt pour une étude des effets des foudroiements de proximité) ;
- les tensions induites sur un certain nombre de boucles disposées à l'intérieur de la cellule, aux endroits où l'avionique doit être située ;
- les champs électro-magnétiques recueillis au moment du foudroiement par les antennes des systèmes de radio-communication et de radio-navigation.

Il est très important de disposer de systèmes de mesure à très large bande (jusqu'à 200 MHz) avec transmission par fibre optique. Par ailleurs, comme les évènements intéressants n'occupent que des créneaux insignifiants du temps total de l'expérience, il est nécessaire de disposer à bord de chaînes élaborées d'acquisition et de traitement des données ; une bonne solution consiste à digitaliser les mesures dans des fenêtres temporelles de durée compatible avec la partie intéressante du phénomène ; le choix de ces fenêtres pourra résulter du passage d'un phénomène choisi au-dessus d'un

seuil donné ; on peut aussi, avantageusement, recourir à la méthode qui consiste à tout enregistrer et à effacer périodiquement la mémoire, sauf si un évènement intéressant s'est présenté dans la séquence enregistrée (système à mémoire tournante) ; cet artifice permet d'étudier les mécanismes précurseurs avant déclenchement d'un effet important.

Etant donné le nombre important des paramètres et le caractère élaboré du traitement, qu'il faut compléter par un dispositif de synchronisation très précis pour dater correctement des phénomènes dont certains ont la dizaine de nanosecondes pour temps caractéristique, on conçoit la complexité des systèmes de traitement de données nécessaires : on est à la limite de ce que la technique peut offrir, dans un domaine où l'appareillage se perfectionne d'une annéee à l'autre. Par ailleurs, on comprend aussi le soin qu'il faut consacrer aux étalonnages préalables en niveau et en fréquence des capteurs, en prenant bien soin d'opérer in situ. Enfin, l'interprétation des résultats n'est pas évidente et pose de délicats problèmes de théorie électro-magnétique ; il s'agit donc encore de recherche, domaine où l'ingénieur doit collaborer étroitement avec le chercheur si l'on veut éviter les conclusions erronées.

Une conséquence évidente de cette complexité est la nécessité absolue d'utiliser des avions très spécialisés, et, d'autre part, l'impossibilité d'aboutir à un résultat en une seule campagne : les essais se déroulent en plusieurs campagnes pendant la saison favorable de plusieurs années consécutives ; la première de ces campagnes sert à mettre au point le système ; elle entraîne des modifications qui sont essayées l'année suivante ; ce n'est souvent qu'au bout de la troisième année que les premiers résultats significatifs sont obtenus.

3) L'avion doit voler au voisinage de systèmes orageux, et doit même pénétrer à la lisière des zones de précipitation si le foudroiement direct est recherché. La configuration la plus favorable à l'analyse est celle qui combine les essais en vol et les mesures au sol : on peut ainsi disposer de stations qui permettent de localiser les coups de foudre de proximité par rapport à l'avion, ce qui permet d'estimer les termes source du rayonnement à partir des mesures sur l'avion et au sol.

4) Le respect de ces conditions est indispensable pour obtenir des renseignements crédibles. Voici quelques résultats préliminaires des campagnes en cours (un plus grand nombre de résultats doit être publié en juin 1983 au cours d'une conférence spécialisée) [17].

- F-106B : de 1980 à 1981, vingt coups de foudre directs enregistrés. Ces essais ont mis en lumière l'importance des résonances dimensionnelles de l'avion, qui amplifient certaines composantes de la gamme HF (7 à 21 MHz). Les mesures, effectuées avec une bande passante de 50 MHz, montrent des fronts de montée de l'ordre de 25 kA par µs. Par ailleurs, certains éclairs comportent jusqu'à une centaine d'impulsions réparties sur un peu moins d'une seconde [18].

- C-130 : de 1979 à 1981, deux coups de foudre directs enregistrés, plusieurs milliers de foudroiements de proximité (dans un rayon de 5 à 35 km) analysés. On déduit de la corrélation entre les champs électriques et magnétiques mesurés dans la phase de propagation du leader que c'est la présence de l'avion qui a déclenché les deux foudroiements directs. Pour les foudroiements de proximité, les formes d'onde observées sur l'avion sont identiques à celles observées au sol. On trouve aussi une amplification par résonance dimensionnelle de composantes HF (2 à 5 MHz). Cependant, ces résonances ne sont pas excitées si le foudroiement est distant de plus de 5 km.

  Les temps de montée des coups en retour varient de plusieurs microsecondes à 100 nanosecondes ; cependant, on note la présence de nombreuses impulsions à front de montée très raide pendant les périodes de pré-claquage. On peut en déduire que le danger pour l'avionique ne provient pas exclusivement des coups en retour [19].

- C-160 : en 1978, seize foudroiements directs. L'équipement de mesure a subi des modifications et les vols de 1981 ont servi à sa mise au point ; aucune conclusion définitive n'a donc pu être publiée à ce jour [20] [21].

L'analyse de ces résultats préliminaires appelle les remarques suivantes :

1) Il ne faut pas attendre de ces expériences une réponse rapide aux questions encore ouvertes sur les caractéristiques cherchées : la réponse viendra à la fin de la période de dépouillement et d'analyse, c'est-à-dire vers le milieu de la présente décennie. Cette lourdeur expérimentale est le prix de la complexité d'expériences destinées à mesurer les paramètres difficilement accessibles d'un phénomène essentiellement fugitif. On tente actuellement de recueillir des mesures complémentaires, mais qui n'ont pas la rigueur des mesures systématiques que nous avons décrites, sur certains avions de ligne, grâce à des "boites noires" qui mesurent les perturbations électromagnétiques avec une bande passante de l'ordre du MHz. Ces mesures complémentaires ne sauraient, à elles seules, donner la réponse cherchée.

2) Les données obtenues par chaque campagne sont trop peu nombreuses et les méthodes de mesure trop délicates pour que les conclusions puissent être tirées rapidement par chaque expérimentateur : seule la confrontation des résultats et des méthodes peut conduire à la constitution d'une banque de données qui puissent être raisonnablement considérée comme valable : il s'agit donc d'un domaine où la coopération est indispensable et où, par ailleurs, elle ne se heurte à aucun obstacle de confidentialité pour

des raisons militaires ou commerciales. Dans ces conditions, cette coopération a pris une dimension internationale aussi bien par l'organisation de Conférences spécialisées que par application d'accords d'échanges d'information et de personnel.

IV - CONCLUSION

L'introduction des commandes de vol électrique (CAG) et la généralisation des revêtements composites rendent particulièrement urgentes les études relatives à la menace que l'électricité atmosphérique fait peser sur le fonctionnement opérationnel des systèmes avioniques.

Parmi ces études, la caractérisation systématique des effets de l'électricité atmosphérique sur les conditions opérationnelles des aéronefs requiert des essais en vol afin de définir les paramètres physiques encore inconnus de la menace. Ces paramètres une fois connus, on peut optimiser les systèmes de protection qui permettront à l'aéronef de voler par tout temps dans les conditions de sécurité requises. L'optimisation demande des études complémentaires (couplage, durcissement, méthodes de test au sol) qui ne sont pas décrites dans cette synthèse.

Les essais en vol nécessaires sont longs et délicats ; ils exigent des avions munis d'une instrumentation très élaborée. Le dépouillement des mesures et l'analyse des résultats doivent faire l'objet de nombreuses confrontations entre équipes de chercheurs : c'est un domaine où la collaboration internationale peut ajouter une grande valeur à la qualité de la recherche.

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ANNEXE I

La campagne de mesure METEOR NF 11 du Centre d'Essais en Vol de Brétigny, a été organisée en 1979 (18 vols) et 1980 (10 vols) sous l'égide du STTE avec la participation du CEAT, de l'Aérospatiale et de l'ONERA.

1 - Instrumentation

Cet appareil était équipé de trois sondes de courant d'impact situées respectivement sur le radôme et sur le bord d'attaque de chaque voilure et de cinq capteurs de champ électrostatique type moulin à champ pour la mesure du champ électrique extérieur et du potentiel de l'avion (voir figure 6).

La possibilité de monter jusqu'à trente et un déperditeurs passifs instrumentés était prévue. Certains vols étaient effectués avec seulement deux déperditeurs destinés à éviter l'apparition de décharges corona sur les pitots.

Le niveau des bruits radioélectriques était relevé dans les gammes de fréquence VLF (10 kHz, navigation OMEGA), MF (350 kHz, radio-compas), HF (20 MHz, radiocommunication), VHF (130 MHz, VOR) et UHF (379 MHz, radionavigation et radiocommunication).

2 - Caractéristiques des programmes de vol

Les vols effectués ont été programmés pour avoir lieu le plus souvent à l'intérieur des nuages. La fréquence d'occurence de l'impact est élevée : 508 minutes d'impact pour 1440 minutes de vol, soit 35 % du temps, au cours des 18 vols de 1979. On a trouvé que le courant d'impact obtenu dans les cirrus (altitude 10-15 km) était moins continu et moins intense que celui mesuré dans les formations convectives plus basses (alticumulus, congestus). La campagne de mesure de 1980 s'est déroulée préférentiellement dans des nuages bas et l'on a obtenu 310 minutes d'impact pour 614 minutes de vol, soit 49 % du temps. Les valeurs des courants d'impact, rapportés à la surface frontale de l'avion, vont de 25 à 60 $\mu A/m^2$ à haute altitude ; ils dépassent à basse altitude 150 $\mu A/m^2$, avec des points autour de 400 $\mu A/m^2$.

3 - Résultats principaux

Le potentiel de l'avion peut atteindre 300 kV. Le champ électrique extérieur peut atteindre 100 kV/m. Il commence à produire des effets de charge par influence à partir de valeurs de l'ordre de 10 kV/m. Le bruit radioélectrique apparaît, malgré la présence des déperditeurs, sous l'effet combiné du potentiel de l'avion et du champ extérieur.

Dans le domaine MF (radio-compas), le niveau de bruit qui atteint 3 mV/m en l'absence de protection est ramené au-dessous de 80 µV/m dans les conditions les plus sévères lorsque l'avion est équipé de déperditeurs passifs. Cependant, l'efficacité de cette protection par déperditeur décroît fortement en présence du champ électrique externe dû à la proximité de nuages chargés électriquement, probablement par suite d'amorçage de décharges corona directement sur la structure.

ANNEXE II

1) F-106B

Le programme de mesures en vol sur la foudre de la NASA, organisé en liaison avec le Centre Technique de la F.A.A., s'est étendu sur une première période de deux ans (étés de 1980 et de 1981). L'avion est un F-106B instrumenté avec des capteurs de champ électrique statique, des capteurs fournissant les dérivées des composantes électriques et magnétiques du champ électromagnétique, et des capteurs de courant (et de dérivée du courant). Ces derniers capteurs sont montés à la base d'une perche montée à l'avant. Le nombre et la position des autres capteurs sont tels qu'ils permettent la mesure de toutes les composantes spatiales des grandeurs étudiées. La mission type de l'appareil consiste à pénétrer dans l'orage pour obtenir un foudroiement direct.

En 1980, l'avion a été foudroyé dix fois pour 69 pénétrations dans l'orage, et, en 1981, dix fois pour 111 pénétrations. Les 20 foudroiements ont été tous obtenus au-dessus de 15 000 pieds : 30 pénétrations à une altitude inférieure n'ont pas été suivies de foudroiements. Sur les 20 coups de foudre directs, 12 ont été obtenus entre 15 000 et 21 000 pieds pour 125 pénétrations, 3 à 25 000 pieds pour 11 pénétrations, et enfin, 5 durant une pénétration unique à 33 000 pieds.

L'ensemble des résultats obtenus indique que ces foudroiements étaient tous d'intensité modeste, le courant maximum mesuré étant de l'ordre de 14 kA [18].

2) C-130

Le programme de mesures en vol sur la foudre de l'USAF s'est étendu sur une première période de trois ans (1979-1981). L'avion est un WC-130 instrumenté avec des capteurs de champ électrique, de champ magnétique et de densité de courant de peau. Ces capteurs sont montés sur le haut du fuselage à l'avant de l'aile, à l'extrémité de l'aile gauche, et sous la queue.

La mission essentielle de l'appareil est l'analyse des effets électromagnétiques des foudroiements de proximité.

La dynamique des chaînes de mesure était prévue pour mesurer les caractéristiques de foudroiements survenant à une distance comprise entre 0,5 et 30 km de l'aéronef. Les mesures étaient déclenchées lorsque la variation du champ électrique dépassait 10^7 V/m/s (augmentation de 10 V/m en 1 µs).

La durée des évènements enregistrés était courte : il s'agit de fenêtres de 40 µs en 1979, de 164 µs en 1980. C'est dire que les données utiles couvrent une durée totale de quelques secondes pour une campagne de vol.

L'avion survolait le site de Devil's Garden, en Floride, à une altitude moyenne de 15 000 pieds, quoique des mesures aient été relevées à 8 000, 5 000 et même 1 500 pieds. Une infrastructure comprenant un shelter d'instrumentation permettait d'effectuer des corrélations avec des mesures au sol (champ électrique, rayonnement VHF).

L'instrumentation embarquée comprenait, outre le radar météo, un "stormscope" capable de localiser grossièrement les foudroiements.

A titre d'exemple, nous décrirons le foudroiement en vol du 26 août 1981, qui a été obtenu lorsque l'avion volait à la vitesse de 333 km/h, à l'altitude de 16 000 pieds, avec une température extérieure de +5°C, sous un cumulo-nimbus (sommet des nuages 30 000 pieds). L'éclair a duré 460 ms, avec une première phase de 30 ms formée d'impulsions durant quelques microsecondes, séparées de quelques dizaines de microsecondes, et une seconde phase séparée de la première par une période calme de 320 ms. Il s'est accompagné d'un bruit violent ; l'équipage, qui a une grande expérience du vol par mauvaises conditions météorologiques, a estimé que l'attachement de la foudre s'était fait de part et d'autre du fuselage [19].

3) TRANSALL C-160

Le programme de mesure en vol du TRANSALL C160-04 du CEV de Brétigny devait s'étendre en principe sur 1978 (expériences de dégrossissage) et sur 1981, après amélioration des chaînes de mesure. En fait, l'avion n'a pratiquement pas volé en 1981, et les seules mesures dont on dispose sont celles de 1978. Comme pour le F106-B, la mission comportait la pénétration en lisière de zone orageuse pour obtenir des foudroiements directs. L'appareil était équipé en 1978 de perches avant et arrière et d'un shunt de mesure du courant, de capteurs de champ magnétique, et de capteurs de courant de peau. La figure 10 montre l'aspect de l'appareil avec la perche avant ; la figure 11 rassemble des clichés montrant différents capteurs dont l'appareil a été équipé après la révision pour la campagne de 1981. Ces capteurs sont répartis sur l'avion comme le montre la figure 12 [20].

L'avion a volé au-dessus de toute la France, à une vitesse comprise entre 170 et 220 noeuds. Au cours de 7 vols (17 heures de vol), on a obtenu 19 foudroiements, à des niveaux de vol compris entre 10 000 et 15 000 pieds ; la température extérieure était comprise entre -5 et 0°C. L'avion pénétrait au plus à 1 ou 2 miles nautiques à l'intérieur du nuage, et éprouvait une turbulence moyenne, accompagnée de précipitation (en général grêle).

Le niveau des foudroiements était plus élevé que pour le F-106B : 10 % des foudroiements dépassaient 50 kA [21].

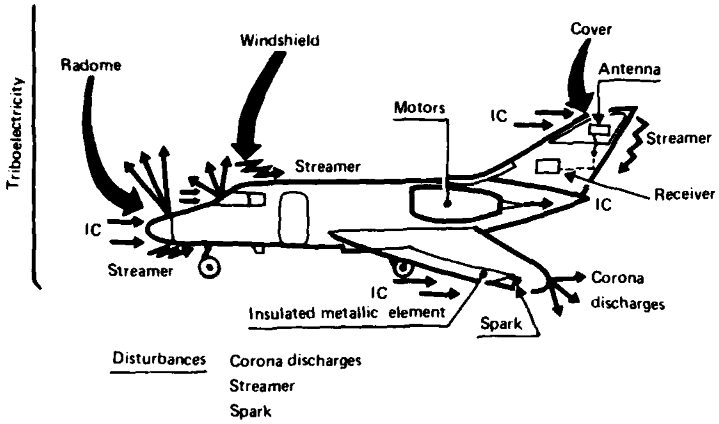


Fig. 1 – Décharges électriques provoquées par l'électricité statique.

Fig. 2 – Décharges de surface (surface streamer).

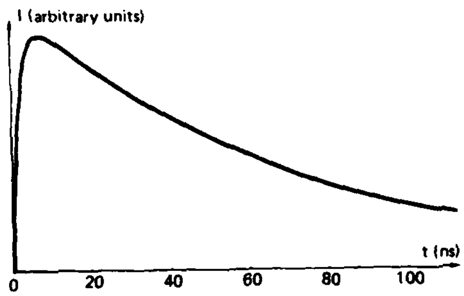


Fig. 3 – Forme typique d'une impulsion de Trichel.

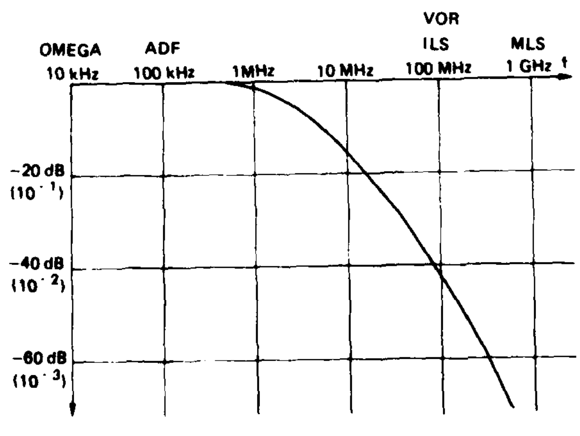


Fig. 4 – Spectre de la puissance rayonnée par la décharge Corona.

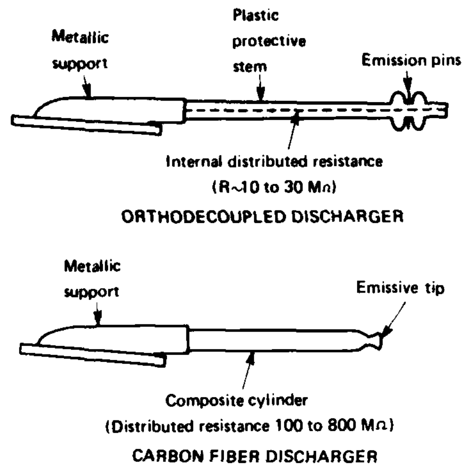


Fig. 5 – Déperditeurs passifs.

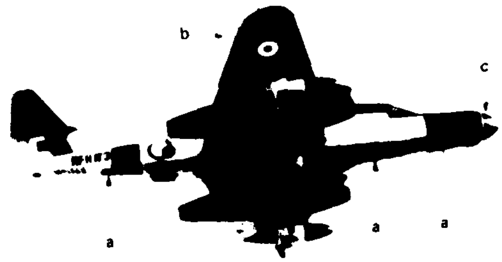


Fig. 6 – Avion METEOR utilisé par le CEV pour l'évaluation en vol des effets des charges électrostatiques.

a) Field mill
b) Passive discharger
c) Impact sensor

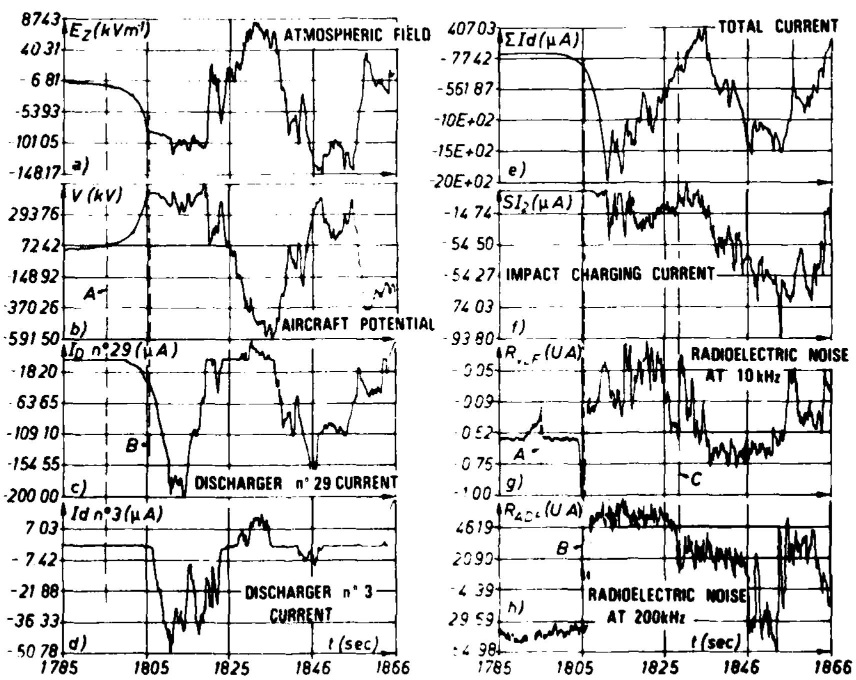


Fig. 7 – Analyse des résultats de la campagne de mesures du METEOR du CEV.

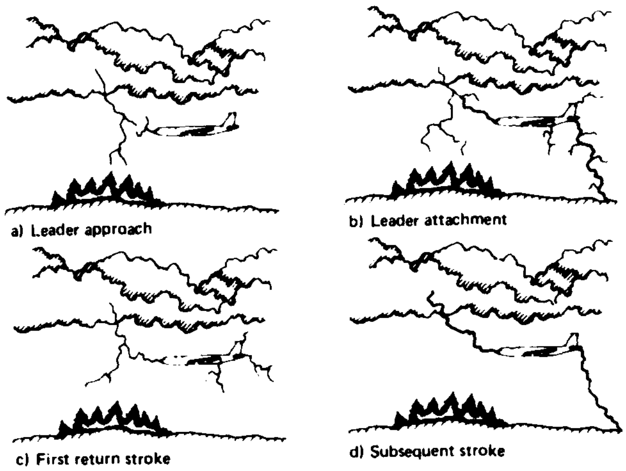


Fig. 8 – Interception de la foudre par un aéronef (d'après [12]).

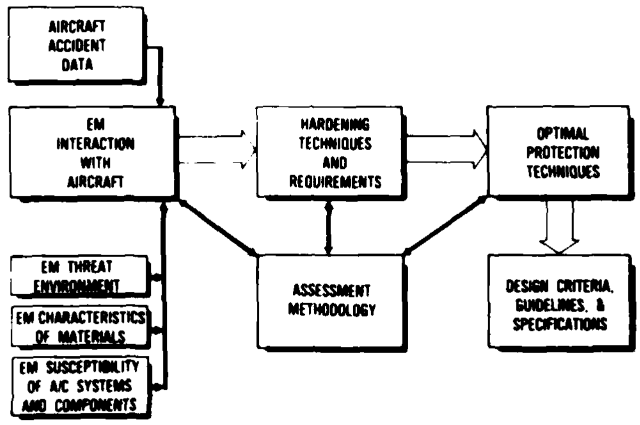


Fig. 9 – Organigramme logique des opérations de recherche et de développement sur la protection des aéronefs contre la foudre (d'après [12]).

Fig. 10 – TRANSALL C-160 du CEV avec sa perche avant.

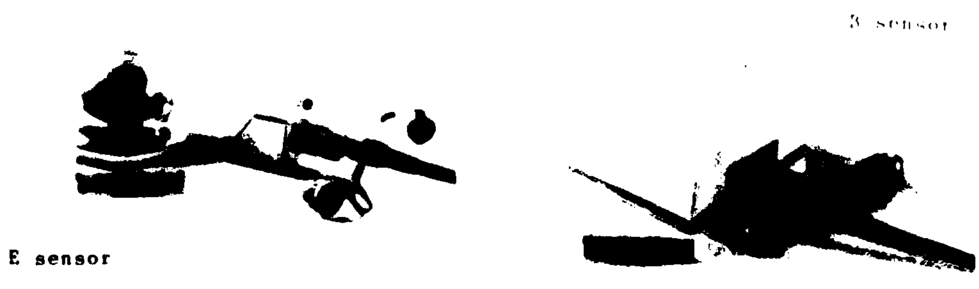
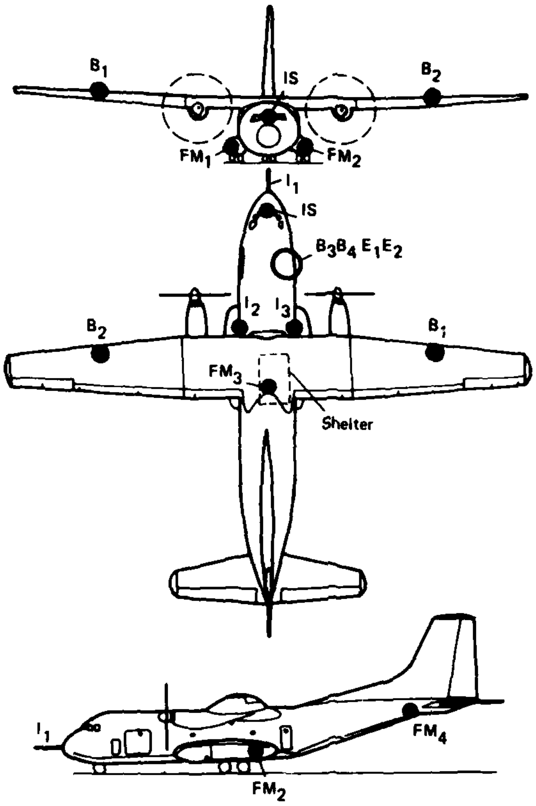


Fig. 11 – Capteurs de mesure du champ électromagnétique montés sur le TRANSALL C-160 du CEV.



FM : field mill — IS : impact sensor
E : E sensor — B : B sensor — il
I 1 : current sensor

Fig. 12 — Répartition des capteurs sur le TRANSALL C-160 du CEV.

WORLDWIDE EXPERIENCE OF WIND SHEAR DURING 1981-1982

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SUMMARY

Large changes of wind and downdraughts (Wind Shears) have caused several major accidents to airliners. The Royal Aircraft Establishment(RAE), with support from the United Kingdom Civil Aviation Authority and British Airways(BA), have analysed wind shear data from over 9000 landings by 26 BA B747 aircraft at 71 airports around the world during 1981 and 1982. The data are analysed using Discrete Gust methods developed at the RAE.

Time histories of wind velocities and aircraft reactions are presented for 9 of the more interesting events identified. These were selected from 86 examples of large wind shears where the three components of wind velocity have been calculated. An example of a severe downburst in the vicinity of a thunderstorm is also presented. This was measured by the RAE HS125 research aircraft during the Joint Airport Weather Studies (JAWS) Programme near Denver, Colorado, USA in July 1982.

Statistics on the probabilities of encountering wind shears with particular patterns of headwind speed changes have been calculated from the 9000 landings. Effects at different airports, height bands, shear lengths, and patterns have been compared. Suggestions are made for Design Cases of shear that could be used for testing autopilots and wind shear measuring systems. Some possible criteria for the severity of single ramp changes in headwind are presented.

1 INTRODUCTION

Large changes of wind and downdraughts (Wind Shears) have been the cause of several major accidents. The worst and one of the most recent accidents occurred during take-off from New Orleans during July 1982 (Ref 1). A Boeing 727 aircraft was taking-off with thunderstorms in the vicinity of the airport and, despite making allowances for a possible encounter with wind shear, the aircraft crashed killing 153 people (Ref 1). Many research activities have been addressing the problems of wind shear since the mid-1970's with particular emphasis on flying techniques and the development of suitable flight deck displays and wind shear sensors. Regulatory Authorities have not been able to use this research to produce practical rules but only advice on flying techniques and ground based wind shear warning systems. This is understandable as it is not possible to produce meaningful rules unless

a) acceptable sensors and/or display systems have been demonstrated, and

b) wind shear design cases have been established for testing such systems.

One of the reasons for the unwillingness to accept various sensors and systems is the scarcity of data from which to provide design cases of wind shear. Such design cases must be realistic in their form with time and position, their size, and their probability of encounter. Until recently the only data available on large wind shears were a few reconstructions from major accidents and limited data from instrumented masts. There were no data on the probabilities of encountering wind shears of a particular form and size, although the loss of 1 or 2 commercial airliners each year through wind shear shows that the probability of encountering a catastrophic wind shear is significant. The need for such data is indicated by the US Federal Aviation Administration's issue of a Draft Advisory Circular (No. 120) giving several examples of wind shear that may be used for testing sensors and displays. However, the examples used are based on the limited data from accidents and masts.

Some four years ago the UK Royal Aircraft Establishment (RAE) with support from the UK Civil Aviation Authority (CAA) approached British Airways (BA) with a view to obtaining wind shear data during landings from the routine flight recorders carried on many of their aircraft. British Airways were interested in obtaining wind shear data to assist in the interpretation of unusual events and a programme was agreed whereby the RAE would provide BA with detailed methods for detecting and analysing headwind components of wind shear (Ref 2) and BA would provide relevant records to the RAE for more detailed analysis. The programme would analyse data from the 26 B747 aircraft in the BA fleet and the aim was to collect the final two minutes of about 12000 landings.

This paper briefly describes the analysis methods used to identify wind shears from over 9000 landings during 1981 and 1982. The paper starts with the scope of the

measurements, then several examples are given of different types of shear, followed by the identification of differences between various airports and the effects of height above the ground. The form of statistical distribution of wind shears for this part of the BA route structure is presented and ways of using these data to provide design cases are described.

2 WIND SHEAR PROGRAMME

2.1 Programme Aims

The programme has three aims:

a) To provide statistics on the probabilities of encountering severe wind shear of various patterns both at individual airports and in a worldwide route structure.

b) To provide a variety of examples of large wind shear to improve the understanding of the different forms of shear and the associated aircraft behaviour.

c) To prove the effectiveness of the Discrete Gust Analysis methods (Ref 3) in detecting wind shears and provide a method for routine application at British Airways.

From these results it is expected that recommendations will be made on the forms and magnitude of various wind shear patterns to be used for wind shear design cases. It is also suggested that the analysis methods are particularly suited to the analysis of wind shear for aviation applications in an economical and effective manner and as such could be used as a general method by other agencies. This would have the advantage that data could be exchanged and compared from a variety of sources. The discrete gust form of the data is particularly relevant to aircraft operations BUT cannot be compared directly with other statistical data.

2.2 Discrete Gust Analysis, or Wind Shear Pattern Detection

Aircraft in flight are unaffected by steady wind velocity except for navigation. An aircraft can perform manoeuvres without considering the wind velocity. However, any CHANGE in wind velocity with respect to the ground with time or position along the flight path will require action from the pilot if the desired flight path is to be achieved. The extent of the pilot's actions will depend on the duration and the total size of the wind velocity changes.

Thus any wind shear (or turbulence) analysis method must identify size and duration of wind changes (NB: It follows from the above that the change of headwind component observed when an aircraft makes a turn in a steady wind is not a wind shear as there is no change of wind velocity with respect to the ground.)

The next essential for wind shear analysis is the ability to detect large isolated events in time histories of the wind. Traditionally Fourier Analysis methods have been used but they suffer from the inherent assuption that the data can be considered as a portion of a regular and continuous stream of data. The problems with isolated events are clear if it is remembered that the Fourier spectrum of an isolated spike is white noise and only by defining the relative phases of each frequency component can the presence of a single spike be detected. Traditionally fluctuating data is often presented as only the power spectrum of the data and it would be impossible to identify a single spike from the power spectrum.

An alternative is to search for particular patterns in the time histories and the Discrete Gust methods developed by J G Jones at the RAE have proved an effective means of analysis over many years of turbulence measurements. Thus it was chosen for the analysis of the BA wind shear data.

The patterns chosen for wind shear analysis are shown in Fig 1 and the analysis programme uses a combination of smoothing filters and differencing across a given time interval to identify the size of patterns of a particular length and shape. As the data are required for defining wind shears that are relevant to a wide range of aircraft the time intervals chosen to define the length of the patterns were 4, 8 and 16 seconds, which correspond to air distances of about 300, 600 and 1200 m at typical approach speeds for the B747. Patterns with a shorter length would be seen as turbulence by most aircraft, even at lower approach speeds, and not require significant pilot intervention to hold the flight path. Patterns with a much greater length will have an extremely low probability of having a very large rate of wind change and thus changes of airspeed will be comfortably within the capabilities of the aircraft and pilot.

The form of analysis identifies patterns within an octave band of lengths where the nominal length is the centre. Thus the total range of lengths covered is from 200 m to 1600 m for the three single ramp patterns and up to three times that for the longest double ramp patterns. Fig 2 shows typical outputs from the analysis programme after filtering and differencing for single ramps and double ramps with zero spacing (ie six of the twelve patterns). Individual wind shears are identified by the peaks and troughs of these filtered time histories. The peak in the time history (Fig 2a) at -17s is correctly identified as an 8s double ramp as it is the largest peak in the filtered

outputs, which is just less than twice the average headwind change. In this example the true average headwind change in the double ramp is 17 kt.

2.3 Analysis of British Airways Landing Data

Initially the flight data are processed at BA to extract headwind, crosswind, aircraft heading, and height at 1 sec intervals during the final 2 min before touchdown. BA processes these data through a simple wind shear identification programme and identifies

a) landings where the magnitude of any shear pattern exceeds a predetermined threshold, which are called Alerts;

b) landings where a combination of wind and aircraft heading change will give a significant false wind shear in the headwind time history. These are not checked for Alerts but the data are passed to the RAE for checking and, if any possible Alerts are present, the data are processed through a more elaborate analysis programme which eliminates false shears arising from heading changes;

c) landings where more than 20% of the data have lost synchronization are rejected at BA.

BA pass headwind and height data for all landings to the RAE, other than those identified with (b) and (c) above. The data are then subjected to a series of tests to identify runs with suspect data. These are then visually inspected together with all wind shears whose magnitudes are above those for a 5% probability of occurrence. Only those landings where the data is almost certainly real wind shear are retained in the statistical data set. However, some landings where corrupt data are well separated from an obvious significant wind shear are analysed to identify the wind shear. These are kept separate from the statistical data. The Alert thresholds are set at wind shear magnitudes which result in about 1.5% of landings containing an Alert.

Landings identified by BA as containing an Alert are transferred to the RAE as a full set of aircraft data (Table 1) which, if the data are shown to be valid, are then analysed to determine a wide range of relevant parameters (Table 1), including all three components of the wind in earth axes.

The headwind shear data are collected by airport and also sorted into height bands between 0, 250, 500, 1000 and 1500 ft, above touchdown. The data are also amalgamated into a total set for the BA worldwide route structure flown by the B747.

Analysis has been completed on 9135 landings recorded between February 1981 and September 1982. This represents about 50% of the total landings by the 26 aircraft of BA's B747 fleet in this period. It should be noted that the analysis programmes use parameters already recorded by BA and the complete analysis has required about 0.5 man-years of software development and about 0.3 man-years of analysis effort at the RAE. Similar software effort was bought in at BA and less than 0.1 man-years were required to run the programmes at BA over the 19 mth period. This very low level of effort is largely due to the effectiveness of the Discrete Gust methods and the care spent in defining the ways of presenting the analysed data before specifying the software requirements. Also, a preliminary set of 1200 landings was analysed (Ref 4) and various improvements to the analysis procedures introduced before proceeding with the larger data set. The main change being the introduction of ways of dealing with false wind shears resulting from heading changes.

Landings have been recorded at over 70 airports around the world and Fig 3 shows the locations of airports where more than 25 landing records have been collected. Details of the numbers of records at each airport are given in Table 2. London (Heathrow) as the hub of the BA network has 26.4% of the landings, 10 other airports have more than 200 landings; 15 have between 100 and 200 landings; 12 have between 50 and 100 landings; 13 have between 25 and 50 landings and 20 have less than 25 landings.

Take-offs have not been analysed because the time spent in the various height bands is much shorter and the speed variation is much greater. Thus it would be difficult to obtain a controlled data set for statistical analysis. The statistical data gathered from landings should be directly relevant to take-off cases as they are expressed in terms of wind changes and airborne distances.

3 LARGE WIND SHEAR

Detailed study of large wind shears serves several purposes, viz.

a) It checks that they are not the result of corrupt data if the aircraft behaviour is consistent with the shear. This is particularly important if the tails of the distribution are to be extrapolated to obtain design cases;

b) it confirms that the Discrete Gust method is identifying the correct magnitude, length and pattern for large shears;

c) it shows the relationship between downdraughts and the various forms of headwind shear so that complete shear models can be derived.

The policy of identifying significant wind shears by their headwind component is based on the fact that significant downdraughts cannot exist near the ground without significant horizontal wind components, and headwind is the easiest of the three components to identify reliably. There is no implication that downdraughts are less significant than headwinds. The severity of a wind shear will depend on the combined effects of both components.

86 Alerts have been analysed to provide three component wind data. Several of these are used in this section to illustrate various features of wind shear. To complete the picture there is also an example of a thunderstorm downburst recorded on the RAE HS125 research aircrft during the Joint Airport Weather Studies (JAWS) research programme near Denver, Colorado, USA in July 1982.

3.1 San Francisco, August 1981

San Francisco is typical of many airports on the BA B747 routes in being close to a large area of water. In this situation it is common to have a cool on-shore wind during the day. Fig 4 shows an example of such a wind. The aircraft (Fig 4a) is not significantly affected although the wind increases by 16 kt in 16s. It is interesting to note the two cool layers identified by the temperature time history in Fig 4b, with a modest low-level jet at the top of the upper layer at about 1500 ft above touchdown. This 'on-shore wind' type of shear was responsible for an accident to a DC10 at Boston when the pilot did not apply thrust early enough when the headwind stopped increasing. The combination of decreasing wind and increasing temperature with height can cause some problems during take-off as the temperature increase reduces thrust at the same time as the headwind decreases. This can also coincide with the reduction of thrust for noise abatement. For this landing the temperature at 2000 ft is nearly 8 deg Celsius warmer than the surface. There are no significant updraughts with this shear.

3.2 Melbourne, September 1981

Melbourne is another coastal airport and the wind shear recorded in September 1981 did cause problems (Fig 5a). At about 50s from touchdown the airspeed fell rapidly by over 20 kt and the rate of descent almost doubled. Thrust was increased almost immediately to a level just greater than that for level flight and the airspeed recovered about half of the loss but the descent rate only slowly reduced towards the normal approach value. The aircraft continued to descend slowly beneath the glide path until at 22s to touchdown it was at 140 ft, which is about 100 ft below the glide path. At this point the increased thrust suddenly became effective and the aircraft was able to regain an acceptable flight path and make a normal landing. The problems arose from a combination of headwind and downdraught variations (Fig 5b). The wind at 1000 ft was 33 kt from about 50 deg to starboard. At 600 ft the aircraft entered the top of an inversion with a downdraught of 1200 ft/min, a wind speed loss of 17 kt and direction changing to 90 deg to starboard. This combination of speed loss and direction change results in a change in headwind component of 29 kt in 20s (a distance of 1740 m), which is a mean rate of 1.5 kt/s. (NB this is the underlying change in the headwind after filtering most of the shorter transients (turbulence).)

At the moment when the headwind decrease stops there is another large downdraught of nearly 900 ft/min and then the shears cease. It is at this point that the aircraft starts to recover.

3.3 Anchorage, March 1981

This was the only wind shear incident reported on the BA B747 fleet in the past two years. The event occurred just before the first landing at Anchorage recorded for the full programme and is not included in the statistical data set. However, a full data listing was obtained and processed through the programme that calculates the three components of the winds. The wind shear programme with BA deliberately did not include any special wind shear reporting, as it is sometimes difficult to sort out wind shear incidents from general comments on less significant shears.

On the first approach the aircraft encountered severe wind shear and overshot. Wind shear was still present 9 minutes later during the second approach but the aircraft was able to land safely. The aircraft behaviour on the first approach is shown in Fig 6a from about 100s (-120s point in Fig 6a) before the start of the overshoot. A normal descent was started at -90s but an excessive sink of 1800 ft/min developed. Thrust was increased at -65s and restored the aircraft almost to level flight. As the glide path is approached the thrust is again reduced (-50s). Again a high sink rate of 1600 ft/min developed and then reduced accompanied by an airspeed loss of 22 kt. At -23s the descent rate again increased to 1600 ft/min and, with the aircraft about 500 ft above touchdown and descending well below the glide path, full thrust was applied for an overshoot. The Ground Proximity Warning System operated a couple of seconds later while the thrust, which lags behind throttle movement, was increasing for the overshoot. After sinking to a minimum height of 420 ft the aircraft climbs away safely.

The winds and temperature (Fig 6b) and the ground track with wind vectors (Fig 6c) complete the history of this event. To avoid high ground, aircraft on the approach to runway 24 have to turn about 90 deg onto the extended runway centreline about 4 nm from touchdown. The aircraft is lined up on the centreline at -50s. The apparent shear in the headwind time history between -80s and -50s arises from the heading change and is

entirely false. This is an example of the false shears that can arise if no special analysis method is used to account for heading changes. The shears between -30s and -20s are genuine as there are negligible heading changes. The very large shear in the total wind when it falls by 35 kt between -55s and -40s has little effect on the aircraft because the change is almost entirely in the crosswind component. The main effects on the aircraft come from updraughts and downdraughts and the headwind changes at about -30s. It is interesting that the updraught at -25s is of warmer air, which probably originated in the warmer layer encountered at about -60s and a few hundred feet higher. The final downdraught which led to the overshoot reached 1100 ft/min and lasted about 12s.

3.4 Harare (Zimbabwe), August 1981

The aircraft behaviour (Fig 7a) shows two large thrust applications at 70s and 30s before touchdown and there is an airspeed increase at 50s. The aircraft sinks below the desired flight path from 45s, despite a thrust increase, and then recovers at 20s.

The main features of the winds and downdraughts (Fig 7b) are the direction change of 180 deg between 42s and 35s and a prolonged updraught of 500 ft/min at about 70s followed by downdraughts of 600 ft/min at 48s and 26s. The first thrust increase is used to correct a low airspeed when the aircraft is low on the flight path. Recovery is assisted by the updraught and thrust is rapidly reduced. Then there is a surge in airspeed at the same time as the first downdraught and the wind direction starts to change. The loss of headwind of 18 kt in 12s, which starts at 45s, and the subsequent downdraught at 26s are the most significant wind shears. They cause the aircraft to sink significantly below the desired flight path. The airspeed surge at 50s is counterbalanced by the downdraught and has little effect on the aircraft.

The air temperature has a peak at the start of the wind direction change which shows that the air at the lower heights is cooler. This and the very large change in wind direction of 180 deg are the sort of pattern which would be found with a storm front.

3.5 Detroit, April 1982

The aircraft behaviour (Fig 8a) shows a particularly disturbed approach with frequent airspeed variations of over 10 kt and descent rate varying between 200 and 1000 ft/min. Then at 28s and 11s there are two dramatic temporary losses of airspeed, although only the second (and less dramatic) results in any significant sink. Thrust changes are few except at the two airspeed losses.

Full analysis is only possible from 113s as the inertial ground speed data were lost briefly from 117s to 114s. This wind shear is an example of one which is not included in the statistical data set but has been analysed to identify large wind shears. Only landings where 120s of uncorrupted data are available have been used for the statistical data set. No air temperature data are available for this landing but the winds are shown in Fig 8b. The wind speed of 50 kt is high and only decays a little from 300 ft down to touchdown.

The various wind components overlap on Fig 8b but it is clear that most of the variations in descent rate from 120s to 40s are caused by vertical draughts. The first of the major double ramp headwind changes at 28s has ramps 6s (476 m) long and an average headwind change of 16.5 kt (2.8 kt/s). It is accompanied by an updraught of 900 ft/min and it is this which alleviates most of the effects of the airspeed loss. Indeed the airspeed changes are exaggerated by the deliberate speed reduction started at 42s and an overshoot in speed as the thrust increase in response to the sudden airspeed reduction during the wind shear takes effect at the same time as the wind increases again. The minimum airspeed was 127 kt, which is 25 kt below the average airspeed for this landing.

The second airspeed loss at 11s is caused by another double ramp headwind variation with 8s (634 m) long ramps and a mean speed change of 13.7 kt (1.7 kt/s). However, this wind shear is not accompanied by any significant vertical wind changes and it does cause the aircraft to sink below the desired flight path. The airspeed falls to 135 kt which is 17 kt below the average for this landing. The incidence angle of the aircraft, which is the only reliable indication of its proximity to stalling, is 6 deg higher than the normal approach value during this second shear. During the first shear it reaches 4 deg higher than normal and, thus, the aircraft was nearer stalling in the second shear although the airspeed was 8 kt higher than during the first shear. This illustrates the fallacy of using airspeed as a measure of the safety margin from the stall in highly dynamic manoeuvering situations such as wind shear encounters. Incidence and related measurements such as those from stall warning sensors are the only relevant indications. Pilots should make maximum use of the lift that is available by increasing pitch attitude if necessary and never attempt to lower the nose to gain speed. But on most aircraft there is no continuous indication of the margin to the stall in dynamic situations such as would be available if incidence were measured. Thus pilots can only use airspeed as a guide and will always be tempted to use every available means to regain normal approach speed as soon as possible. Advice such as flying at the stall warning (stick shaker or natural buffet) cannot be applied in practice because the pilot has no information on his rate of approach to stall warning, of his margin from it, or degree of penetration beyond it.

The wind shears of this example also indicate the importance of the timing of reversals, which is the reason behind the inclusion of double ramp patterns with different time intervals between the ramps in the patterns identified by this programme. In this landing the phasing of thrust with the wind shear increases both the maximum and minimum airspeeds by 8-10 kt compared with the sizes of the wind shears.

Disturbances of the kind seen in this landing at heights below 300 ft and in strong winds can be caused by the wakes of large buildings on or near the airport. There are no particular warnings in the landing charts for Detroit on Runways 21L or 21R, but if this landing was on 21L then the aircraft would be in the lee of the central terminal buildings at the place where the large shears were encountered. It is not possible to differentiate between runways with the same heading from the records as only the runway heading is known.

3.6 Muscat, June 1982

There is a tendency to worry most about wind shears that cause a loss of height because the consequences of landing short of the runway are often catastrophic. But it is possible to cause major damage by landing too far along the runway and overrunning the end even if the majority of the passengers escape without injury. This wind shear at Muscat on the Gulf of Oman in the Middle East demonstrates a type of wind shear which could cause an overrun accident although in this case there was sufficient runway available for a safe landing. The aircraft (Fig 9a) made a steady approach and starts the landing flare at 20s and should have landed at about 15s. Instead the aircraft floats for about 1 km (13s) even though the thrust has been reduced to idle.

The winds (Fig 9b) show that, although the wind speed did not vary significantly, the wind direction swung through about 150 deg to a tailwind between 38s and 14s and then back through 170 deg to a headwind as the aircraft flared for landing. The whole event is a wind shear with a double ramp pattern with a 16s (1210 m) ramp length, an interval of 16s and a mean wind change of 13.3 kt. The headwind increase that causes the float is part of this double ramp and has a 13 kt increase in 12s (1.1 kt/s). This just about balances the normal deceleration in airspeed after reducing thrust for touchdown.

3.7 Chicago (O'Hare), April 1982

Just as a rapid increase in headwind during the landing flare can cause a long float, so a rapid decrease can be one of the causes of a heavy landing. In this landing at Chicago the aircraft (Fig 10a) made a reasonably steady approach with few thrust changes except to correct the low airspeed at 30s. At 8s the airspeed starts to drop rapidly and, by the time the thrust reduction for the landing flare starts, it has fallen about 12 kt. It should be remembered that the thrust lags the pilots throttle movement. The consequence of the loss of airspeed is that the descent rate does not reduce during the flare as it normally would (see the Detroit landing, Fig 8a). The rate of descent at touchdown on this Chicago landing was 10 ft/s and this gave a vertical acceleration increment of 0.56 g. This is a hard, but not quite a heavy, landing.

The winds (Fig 10b) varied quite a bit during the early part of the approach, but were fairly steady from 50s until 5s. At 5s the wind drops by 13 kt in 4s (3.3 kt/s). At this point the aircraft is about 70 ft above the ground and starting the landing flare and it is too late for any thrust increase to take effect before the aircraft lands. In any case there is not sufficient excess thrust available to provide 3.3 kt/s acceleration when the aircraft is in the landing configuration with flaps and undercarriage down.

Several other hard landings as a result of a wind shear during the flare have been seen among the B747 data, but it is not the only cause of hard landings.

3.8 Anchorage, November 1981

It has been suggested that a large difference of, say, 25 kt between the headwind during an approach and that quoted for touchdown can be used as a warning of potentially serious wind shear. Of the examples presented in this section only the wind shear during the landing at Harare (Fig 7b) would be identified by this criterion at the start of the approach, and the shear at Melbourne (Fig 5b) would be identified in time to warn the pilot. The other shears would not be identified.

There are also quite a few occasions, such as this landing at Anchorage (Figs 11a and b) where a strong headwind at the start of an approach decays quite steadily to a light wind at touchdown. In this case from 40 kt at 65s (about 800 ft above touchdown) to 5 kt at 15s (about 100 ft above touchdown). This decrease of 0.7 kt/s is easily countered by a moderate thrust increase once the decreasing headwind is recognized. The approach is generally uneventful apart from the effects of the vertical winds between 40s and 30s.

The headwind change criterion is still useful but it should be noted that it will not identify all large wind shears and it will identify quite a few fairly innocuous shears. It can do little more than warn a pilot that there is a possibility of encountering a large shear.

3.9 London (Heathrow), April 1982

The final example chosen from the BA B747 records is rather unusual. It shows (Fig 12a) an uneventful approach with a sudden spike in airspeed at 10s and a small spike on the pressure altitude at the same time. The winds in Fig 12b show vertical wind and wind direction changes as well as a horizontal wind spike of 11 kt at 10s. All these wind changes and the brief reduction in static pressure shown on the pressure altitude time history are characteristic of an encounter with a vortex from a previous aircraft. This is quite likely in the light winds with a small crosswind component and final confirmation comes from the roll attitude time history (not shown in Fig 12) which shows that the aircraft rolls nearly 10 deg to starboard at 10s.

3.10 Thunderstorm Downburst Near Denver, July 1982

Although the majority of the data in this paper are from BA B747 flight records, there is no obvious example of the particularly hazardous downbursts associated with thunderstorm activity. As stated in Section 3, the RAE HS125 research aircraft joined with several aircraft from research agencies in the United States of America and a comprehensive array of ground based equipment in the Joint Airport Weather Studies (JAWS) Project (Ref 5) at Denver, Colorado in the Summer of 1982. the object of the programme was to study large wind shears of various kinds that are associated with thunderstorm activity. These vary from general flow circulations around the storm, such as storm fronts, to very localized events, such as downbursts and tornados. Analysis of the data from a large number of events is proceeding in the USA and at the RAE in the United Kingdom.

An example of a time history of a penetration of a downburst (or, because it is less than 4 km across, a microburst) is shown in Fig 13. The penetration was made at a height of about 1000 ft above ground (Denver is over 5000 ft above sea level) and at a safe speed of about 210 kt CAS. The microburst is about 2.4 km across and the maximum headwind change is -35 kt in 5.3s (640 m). The downdraught fluctuates considerably and requires a mean change of +3 deg in pitch attitude to maintain the flight path. This corresponds to a mean downdraught of about 1200 ft/min which lasts for about 5s, or a distance of about 600 m. The fluctuations in downdraught produced +/- 1'g' excursions in normal acceleration at the flight speed of 210 kt CAS, which would be reduced to about +/- 0.6 'g' at normal approach speeds.

These microbursts occurred quite frequently in the vicinity of thunderstorms in the Denver area and could be associated with rain or only virga (light rain which evaporates before reaching the ground). They usually built up and then decayed within 5 to 15 min and as one died another would often occur a few kilometers aways. The 'dry' microbursts under virga could be seen quite clearly in the countryside around Denver by the curtain of dust rising to over 1000 ft around the perimeter at a time when the winds outside the microburst were light. This curtain of dust must be carried on a strong updraught and this is not compatible with the model of downbursts as a vertical jet. A vertical jet impinging on the ground will produce a strong outflow with only weak buoyancy forces to produce any updraught. Thus the dust should blow out from the centre as an almost horizontal sheet. Following the observations of the JAWS programme it has been suggested by Caracena at a Wind Shear Mini-Workshop at the University of Tennessee Space Institute in October 1982 and by Fujita (Ref 6) in evidence to the enquiry into the New Orleans accident (Ref 1) that microbusts are the flow around a horizontal vortex ring. This explains the dust curtains and the way the energetic flow velocities are contained within such a small area. It would also produce strong downdraughts close to the ground and require less energy than a jet with a similar downdraught. The data of Fig 13 would correspond best to a model with several vortex rings being produced one after the other (Fig 14) and the HS125 passing between two rings. The lower vortex would produce the loss of headwind at 36s and the increase at 54s and the upper vortex would cause the main headwind changes as the aircraft appears to have passed closer to the core of the upper vortex ring.

It seems that the microburst could be caused by localized pulsation in or just below the cloud. The suggestion that the energy changes needed to produce a microburst comes from the evaporation of cloud moisture is supported by the light patch visible in the cloud after a microburst decays.

Although microbursts are quite common, their short duration and small size mean that the probability of an aircraft encountering one at its peak strength on approach or take-off at an airport is rare, but, as several major accidents show, not insignificant. These same characteristics also mean that any ground based equipment that is going to detect the growth of a microburst should scan the complete area in less than about 3 min and should have a separation between measuring points which is not more than about 1 km.

4 WIND SHEAR STATISTICS

The statistics from 9135 landings represent over a million seconds, or over 300 hours of records covering an airborne distance of about 83000 km which is more than twice round the world. They have been used to identify both the differences between various groupings of the data, eg airports and height bands, and also the overall characteristics of the distribution of the data so that extrapolation to design cases can be made with an acceptable level of confidence. The main reasons for collecting such a large data sample are to have sufficient examples of large wind shears to establish

a) that they are part of the general distribution of all wind shears both large and small, and not part of a different distribution of large shears;

b) the form of the distribution, which is not expected to be Gaussian. About 100000 data points are required to define the distribution of the tail with reasonable confidence.

Ideally more data would be desirable, but to increase it by a further factor of 10 would require 20 years of analysis at the present rate. As the distributions contain over 150 large wind shears and are well defined, the statistical analysis has been completed. Analysis of large events will however continue at the CAA and BA as part of the CAADRP programme.

4.1 Presentation of Data

Wind shears are identified with the patterns shown in Fig 1 and the measured values of headwind change. Basic ramp lengths were chosen in octave bands about the nominal values for 4, 8 and 16s (approximately 300, 600 and 1200 m). This means that data for nominal ramp lengths of 300, 600 and 1200 m covers ramps in the ranges 200-400 m, 400-800 m and 800-1600 m respectively. To simplify the interpretation of the data and its application to ramp patterns of a particular length it is very desirable to find a way of normalizing the data so that it collapses onto a single curve for all ramp lengths. One way of nomalizing the ramp pattern data is that used by J G Jones (Ref 3) for turbulence data. The number of shears is divided by the total number of ramp lengths travelled and the 'speed change' divided by (ramp length)$^{1/3}$. Strictly the (ramp length)$^{1/3}$ factor should only collapse the data for ramp lengths less than the scale length as it relates to the region of the power spectrum of atmospheric turbulence with a slope of -5/3. Scale length is defined as the longest wavelength where the slope of the power spectrum is -5/3. Wind shears have long wavelengths which may be greater than the scale length. However, it was decided to use the same normalization process for wind shear data to provide a convenient basis for comparison with turbulence data and because it may prove adequate for normalizing much of the data.

The effect of using this normalizing method on data for single ramps with decreasing winds from 244 landings at New York (JFK) are shown in Fig 15. The data are presented as cumulative distributions for three nominal ramp lengths of 299, 599 and 1198 m. In a typical cumulative distribution the number of occurrences where the headwind change exceeds a given level are divided by the total time of the sample and plotted against the headwind change value, ie it is the integral of the frequency distribution from that headwind value to infinity with positive and negative speed changes considered separately. In Fig 15, for example, there are 0.1 exceedances of -7 kt (ie more negative than - 7 kt) per minute, which means one would expect to encounter a headwind decrease of more than 7 kt about once in every 10 min on the approach to JFK. At JFK the normalizing factors are very effective in collapsing data for the three ramp lengths onto a single curve. This is found to be the case for quite a lot of airports. The divergence at the tails is because there are insufficient events to establish the curves with any confidence.

All the data in this section (Section 4) are presented in this normalized form of cumulative distributions ('exceedances'). On many of the figures the shear pattern and nominal ramp length are shown by a small diagram of speed against distance similar to those in Fig 1.

In the following sections it will be necessary to cross refer to various figures and the normal process of numbering figures sequentially at their first appearance will not produce a logical sequence of figures for the reader who is not studying the complete text. Thus the figures are presented in the sequence most appropriate to the order of the sections.

4.2 Overall Probabilities

This section takes all the data from all airports and looks at the probabilities of a BA B747 encountering particular wind shears during normal operations. Because of the wide range of airports served, these probabilities may well be representative of those for other airlines on worldwide routes. They are not necessarily representative of the probabilities for local routes such as those within Europe or mainland USA. However, data on probabilities of encounter at individual airports (see Section 4.3) will be relevant to those situations.

Three main aspects of the overall statistics are considered in this paper. They are

a) different ramp lengths;

b) different ramp patterns, and

c) different height bands.

4.2.1. Single ramp patterns of different lengths

Cumulative distributions of single ramps with nominal lengths of 305, 609 and 1218 m are shown in Fig 16. The 609 m and 1218 m data collapse well onto a common line but

there are more 305 m ramps than would be needed for such a collapse. This is shown later (Fig 24) to be a situation that is peculiar to certain airports and, as these include London (LHR) which has 26% of all the landings, the separate line for 305m ramps is only going to be relevant for route systems whose hub airport has this characteristic. There are several airports with this characteristic (Section 4.3.2.3) but London shows it most strongly. Thus it is likely that the 305 m ramps will show the same tendency for other worldwide route structures, but perhaps less marked than in these data.

The most important features of Fig 16 are the linearity of the cumulative distributions at large wind shear values. This will provide a good basis for extrapolation of the results to define suitable values for design cases. The deviations from linearity at the lowest probabilities are not significant. They represent a few isolated events and thus their true probability of occurring is not known.

The collapse of the data with ramp length is similar for all ramp patterns and for most of the data in the next few sections only that at a nominal ramp length of about 600 m is presented.

The slight spread of exceedances at the origin for increasing headwinds is due to the mean gradient of headwind with height and is considered in more detail in Section 4.2.3.

4.2.2 Different shear patterns with a nominal length of 600 m

Fig 17 shows cumulative probability curves for all four shear patterns at a nominal ramp length of 609 m for all landings. The speed change scale can be presented on this figure as only one ramp length is considered. One of the main features is the very small difference in probability between all three types of double ramp shears. As may be expected, the double ramp with an interval of one ramp length between the shears is slightly less likely at a given shear size but the differences are small. The other main feature is the almost constant ratio between the speed changes for single ramps and those for double ramps at a given number of exceedances per ramp length. The ratios are approximately 0.74 for decreasing speeds and 0.82 for increasing speeds. The differences between speeds increasing and decreasing are largely due to the biasing effects of the average rate of change and curvature of wind speed with height which is discussed further in the next section (Section 4.2.3.)

With this well defined relationship between all the double shear patterns and single ramp shears it is possible to describe characteristics of all these shear patterns for the overall route network by studying the trends in single ramp shears. However, in some instances, such as studying the character of shears at individual airports, it is wise to consider the simple double ramp pattern as well as single ramps in case there is variation in the relationship because of some local features at particular airports.

4.2.3 Different height bands

Fig 18 shows the cumulative probabilities for single ramps with a nominal length of 608 m separated into four nominal height bands. The separation is actually into time bands but, as most of the landings follow a glide path angle of about 3 deg the time bands can be converted directly into heights. The bands are 0-250, 250-500, 500-1000 and 1000-1500 ft. Zero is taken as the height indicated at touchdown, which is about 40 ft above ground level for the B747. There are no significant changes in the slopes of the data in Fig 18 in different height bands or between increasing or decreasing speed changes. There are, however, shifts in the intercepts at zero speed change, ie the total number of shears. Speed reductions become more probable in the lower height bands. This effect is seen more clearly in Fig 19 where data for all three ramp lengths is compared for the lowest two height bands. The difference in 'speed change' between the increasing and decreasing ramps at a given level of exceedances is due largely to the mean rate of change of headwind with height (the wind gradient). Because of the descending flight path the total probability distribution, which includes both positive and negative 'speed changes', will have a mean which is nearly equal to the gradient of the 'mean headwind' with height multiplied by the change of height in travelling one ramp length. Thus the difference between 'speed changes' at a constant cumulative probability will be twice the change of 'mean headwind' in travelling one ramp length. The inset table on Fig 19 gives the expected gradient of 'mean headwind' in the two lowest height bands derived from data at different ramp lengths. The values agree with each other but are slightly larger than those on the variation of the 'mean headwind' with height (Fig 20) (NB: The height in Fig 20 is taken as 40 ft at touchdown to give height above ground rather than height above touchdown. This is to allow the data to be compared with standard profiles of wind against height). Integrating the gradients derived from the cumulative probability data gives a total speed change over 1500 ft height of 8.2 kt, cf an actual change of 6.9 kt. The difference is due to a small skew in the data between increasing and decreasing speeds. This is shown in Fig 19 by the slightly more negative slope of the decreasing ramps.

Thus the only significant effect of height on wind shear is this bias due to the gradient of the mean wind with height. The bias is quite significant in the last 200 ft. One important consequence of this finding is that it is possible to construct probability distributions appropriate to take-off cases simply by applying biases from the data of Fig 20 in the reverse sense to the landing cases analysed here. In the take-off cases the bias will be positive and, because of the steeper climb gradient, slightly larger.

This will make speed decreases of any given size less likely during take-off than during landing.

Again although there is no particular variation in probabilities of encounter with height other than the bias for the overall data, there may be particular features at individual airports. The data for San Francisco (Fig 22), which will be discussed in Section 4.3.2.1, shows more headwind increases at height between 250 and 500 ft than in other bands. These correspond to the 'on-shore' winds such as that in Fig 4. A similar peculiarity is apparent in 300 m increasing double ramps at Hong Kong (Fig 23a) which correspond to the spike shown in Fig 23b (Fig 6 of Ref 4).

A corollary of this effect of height is that wind shears at a constant height above the ground would be independent of that height. At first glance this conflicts with results from turbulence measurements close to the ground (Ref 8). However, when flying over undulating terrain, the height above ground is not constant and, in increasing and decreasing about the mean height above ground, both positive and negative shears will be added to the distribution. The size of these additional shears will be greater close to the ground and increase with increasing variations in terrain height. However, over flat terrain there should be no effect of altitude. Another difference between landings (and take-offs) and general flight at low altitude is the bias of the flight direction into wind for landings. Finally, turbulence data has, so far, only been presented for individual flights in particular conditions and not for an accumulation of conditions as with the wind shear data. Thus, it is not surprising that there are some apparent differences. It will be interesting to see if further analysis supports the conclusions of this wind shear data.

4.3 Airport Characteristics

The distribution of wind shears at individual airports can be influenced by a wide variety of features such as local topography, season, wind direction, time of day,nearby large areas of water, airport buildings, etc. With a limited sample of data it is not possible to study all these effects and in the examples presented here the data are for the 2 years of the study and all runways, except at Hong Kong where the 50 deg heading change on the approach to Runway 13 prevented any analysis and only data for approaches to the other end of the single runway (Runway 31) are available. The times of day are usually restricted to a narrow band defined by the timetable for the route and the range of landing times is given in Table 2.

Data has been collected in groups of 50 landings at each airport (100 landings at London (LHR)) so that some indication of seasonal variations could be otained at the most frequently visited airports. Aircraft heading after touchdown is available to help identify the runway in use, but with the limited sample size it is not possible to obtain much useful information on the characteristics of the approaches to individual runways, except when studying any large events.

4.3.1 Differences between airports

Fig 21 shows the distribution of negative single ramp and positive double ramp shears of 600 m at 8 airports. At any level of exceedance the airport with the largest wind shears has speed changes of no more than about twice that of the airport with the smallest shears. The lowest shear levels among these airports were at Nairobi (NBO), Kuala Lumpur (KUL) and Singapore (SIN). Landings at NBO are mainly just after sunrise when weather activity is often at it quietest. KUL and SIN on the other hand have landings during the late afternoon and are also renowned for their levels of thunderstorm activity during the summer.

The largest shear levels were at Hong Kong (HKG, RW 31 only), New York (JFK) and London (LHR) for single ramps and Hong Kong (HKG RW 31 only) for double ramps. Hong Kong is surrounded by rugged mountainous terrain and is well known for the high level of turbulence on the approach. Only approaches to Runway 31 could be analysed because of the offset Instrument Landing System and the late heading change of 50 deg required for landings on Runway 13. In general the shape of the distributions are well established, even with only just over 100 landings at an airport. The large event in the distribution for double ramps at San Francisco (SFO) is expected to become part of the general pattern if a larger sample is taken.

Thus, as the hub airport, LHR, is one of the airports with large wind shears and as the overall selection of airports is representative of a wide variety of conditions around the world, the overall distributions of Section 4.2 are believed to be representative of most international route systems.

4.3.2. Individual airport characteristics

From the distribution of wind shears in different height bands and patterns it is often possible to identify the major characteristics of wind shears at individual airports.

4.3.2.1. San Francisco (SFO)

In Fig 22 the single ramp distributions show that the level of positive single ramps between heights of 250 and 500 ft are nearly twice as large as those at other

heights. The difference between positive and negative ramps in this height band shows that the mean wind increases with decreasing height instead of decreasing as it usually does. This behaviour is caused by the frequent on-shore winds of the type shown in Fig 4. Double ramps show no unusual features other than the very large single event with an average speed change of 17 kt. As with all statistical samples, any large event can occur at any time and will only find its true probability level if the sample is very large. Thus single large events will often be displaced from the true probability distribution in this way.

4.3.2.2. Hong Kong (HKG), Runway 31 only

The approach to runway 31 at Hong Kong Airport is over the sea and passes between Hong Kong island and the mainland; both of which are mountainous regions. This approach, is often very turbulent, as indeed is the other approach to runway 13, which could not be analysed because of the 50 degree turn shortly before touchdown. The distribution of wind shear, Fig 23(a) shows slightly larger levels of single ramp shears than at San Francisco. However, the double ramp shears of 305 m, which are nearest to turbulence, have a magnitude almost twice that of the San Francisco double ramp shears. Fig 23(b) shows an example, from Ref 4, of turbulence at Hong Kong. The large double ramp of 4 second ramp length (c 300 m) is fairly common in westerly winds, but does not produce such a distinction shift as the on-shore winds at San Francisco, Fig 22.

4.3.2.3. London (LHR)

In Fig 24 the data for negative single ramps of 301, 602 and 1203 m in the four height bands shows a reasonable collapse of the distributions of the two longer ramps but larger levels of speed changes for the 301 m data. This effect was noted in the preliminary data of Ref 4 where it was speculated that it could be caused by higher levels of turbulence from nearby buildings. A similar effect was noted in Johannesburg (JNB) and Los Angeles (LAX) and is still noticeable in the present data. However, the presence of the larger levels of speed changes at 301 m in all the height bands is surprising as one might expect building turbulence to be more prominent in the lower height bands. The data suggest that (ramp length) rather than $(\text{ramp length})^{1/3}$ will be needed to collapse the data at different lengths. This could imply that the scale length of turbulence is significantly shorter at these airports than it is at many others.

However, whatever the cause, the consequence is that these airports will pose more of a wind shear problem to aircraft with low approach speeds, which are more sensitive to the shorter length shears, than to faster jet transport aircraft, which will mainly notice more turbulence.

4.3.2.4. Kuala Lumpur (KUL)

In contrast to the previous section the data for Kuala Lumpur in Fig 25 shows no unusual features and an excellent collapse of data at all ramp lengths.

5 WIND SHEAR SEVERITY

Various attempts have been made to define a wind shear severity scale to relate wind speed changes and rates of change to the potential hazard to aircraft. One such scale was proposed to the International Civil Aviation Organization (ICAO) and suggested that the changes of wind in 600 m horizontal distance or 30 m vertically should be used to define severity for headwind changes. Table 3 gives details of the proposals presented to ICAO. In practice aircraft are going to be affected by two main factors. First by the rate of change of wind speed in relation to the acceleration that can be achieved with the excess thrust available, which is typically about 3 kt/s for a jet transport aircraft in the approach configuration. Second by the total magnitude of that wind speed change especially relative to the speed margin from the level flight stalling speed, which is the lowest speed at which level flight can be maintained and is typically about 20% (or about 25 kt for a jet transport aircraft). In general wind shear will only be a problem if both these values are significant. This can be seen if one considers two extremes. In one case it is clear that an aircraft can easily respond to even a 100 kt change of wind if the rate of change is only 0.1 kt/s, and similarly for a case at the other extreme with only a 5 kt change, even a rate of change of 10 kt/s is no problem. The use of a fixed distance in association with total wind changes restricts the description of wind shears. For example, the shear at Melbourne (Fig 5) is only 12 kt over 600 m yet it is the total change of 29 kt that is significant. If the shear had only lasted for 600 m then the total flight path deviation would have been much less.

Other problems in defining simple wind shear severity scales are the different responses to single ramps and double ramps. With double ramps the wind eventually returns to nearly the initial value and thus aircraft throttle action is less critical and even unneccessary for short ramp lengths. With single ramps some throttle action is essential to stabilize at the new condition.

However, it is very desirable to give some guidance on wind shear severity to help pilots, meteorological forecasters and aircraft and avionic equipment designers in their judgement of wind shear. The data obtained from the BA flight records give an opportunity to try and suggest a broad guide to wind shear severity, at least for single ramp shears. If the rate of change of headwind and the ratio of its total change to the aircraft's normal approach speed are taken as the main factors and the total change is more significant than rate of change, then a severity factor could be

$$S = (dV/dt).[\text{del } V/Vapp]^2 = [\text{ del } V/(\text{Ramp length})^{1/3}]^3/ Vapp$$

where S = wind shear severity factor

dV/dt = rate of change of wind speed

del V = total change of wind

Vapp = normal approach speed

Thus, the primary parameter turns out to be the normalized wind shear value, [del V/(Ramp length)$^{1/3}$], used in all the statistical data figures.

The relationships between the severity factor, S and ramp length, dV/dt and (del V / Vapp) are shown in Fig 26a. Further consideration of aircraft response suggests that when the rate of change of wind speed is greater than the acceleration available by applying full thrust then it will be the total change of wind that dominates the response of the aircraft, and conversely when the rate of change is low then the total change will not be significant. This suggests that at (dV/dt) $>$ 1.5 m/s^2), which is about the maximum available on transport aircraft in the approach configuration, the lines of equal severity will tend to become parallel to the lines of constant (del V/Vapp) and at the right hand side of the figure the lines of equal severity will be parallel to the lines of constant (dV/dt). Between these two regions it may not be unreasonable to have lines of constant severity at constant values of the severity factor, ie horizontal. Suggestions for possible boundaries are shown in Fig 26b, where the categories of light to severe are the same level of aircraft response used in Table 3.

The boundaries have been based on the BA B747 measurements and Fig 26c shows these with wind shear and ramp length scales for a B747 together with those points for the large wind shears of Figs 4-12 where a single ramp headwind variation was a significant factor. As well as relating the severity parameter to these, a line giving the level of wind shear expected once in 10^7 for the B747 scales is included. This suggests that the probability of a B747 meeting a severe shear in headwind on the British Airways routes is about 2 in 10^6 landings. It should be remembered that this does not include the probability of encountering a severe downdraught. Aircraft with lower approach speeds will be more likely to encounter severe wind shears if the criteria of Fig 26 are relevant.

As the values of the normalized wind shear are constant for a wide range of ramp lengths from about 400 m to 2000 m, it is possible to produce a simplified severity scale and this is shown in Fig 27 with the mean cumulative B747 data. Also shown is the scale proposed to ICAO and it can be seen that that such a scale would have expected a probability of encountering severe shears of about 1 in 700 landings, or about 13 hazardous events in the sample of over 9000 landings from BA B747's. At most only one such event (Fig 5) was found and that was on the borderline between 'strong' and 'severe'. (NB the overshoot event of Fig 6 was caused by downdraughts and is not covered by this headwind criterion.) Thus it is clear that the scale proposed to ICAO was not relevant to B747 operations and does not take sufficient account of the wide range of ramp lengths that are significant to particular aircraft.

However, it must be emphasized that the scales of Fig 26 are only intended as a guide and relate primarily to approach conditions. During take-off the climb angle represents an acceleration capability even though there is no additional thrust available from the engines, other than the use of any emergency ratings, thus the criteria of Fig 26 may still be relevant with the climb speed used instead of Vapp. In the climb the speed margin from the stall is usually slightly smaller but the probability of encountering headwind reductions is less because of the favourable gradient of mean headwind when climbing.

Any severity scale must be tested in piloted flight simulators and stand the test when applied to real flight situations.

6 COMMENTS ON WIND SHEAR DESIGN CASE SIMULATION

One of the main aims of this programme has been to provide data from which realistic design cases can be chosen to test aircraft control and display systems, and detection and control/display systems designed to respond to wind shear. The choice of the levels of probability of encounter is a matter for certificating authorisation and will not be addressed in this report. This section demonstrates how the data may be used to obtain design cases and comments on the relevant levels of turbulence which must be added to the wind shear. Some adequate representation of turbulence is essential because of the difficult compromise needed between fast and false responses. It is recommended that a turbulence model based on Discrete Gust methods, such as that of Ref 8, should be used, as it has a realistic form with more larger and fewer moderate inputs than those models based on white noise.

6.1 Using Discrete Gust Data for Design Cases

To demonstrate the application of the statistical data gathered from the Discrete Gust analysis methods consider the answers to two typical questions:

a) What are the largest speed decreases and their corresponding negative rates of change that are likely to be encountered once in 1000 landings worldwide?

b) At San Francisco, how frequently are on-shore winds with a wind change of +20 kt at +3 kt/100 m likely to be encountered?

For the first question the initial step is to convert 1000 landings into the number of ramp lengths travelled. The average distance travelled in 120s before touchdown at an average true airspeed of 76 m/s is 9.12 Km. Thus the number of ramp lengths in 1000 landing records is 1000* (9120/(Ramp Length)), where (Ramp Length) is the nominal value at the centre of the octave band of (Ramp Length) represented by a single line on cumulative probability plots. Fig 16 gives data for single ramps for the worldwide sample and Table 4 lists the answers to the question for ramps from 200 m, where the (Speed change) is -11.8 kt at -5.9 kt/100 m, up to 1600 m, where the (speed change) is -17.7 kt at -1.1 kt/100 m. Ramp characteristics where the octave bands meet are not identical, although the differences are small when the lines on the normalized cumulative distribution collapse to a single line for different nominal ramp lengths. When they do not collapse then a plot of equi-probable ramp characteristics can be derived by smoothing the normalized data.

The single ramp cases for the rarer probability of 1 in 10,000,000 landings are shown on Fig 26(c). The 1 in 1000 cases approximately correspond to the boundary between 'Moderate' and 'Strong' for B747 on the scale suggested in Fig 26.

In discussing a possible severity scale in Section 5 the need to consider both the (speed change) and the gradient was established and the answer to question a) shows that there is a range of different ramps that are equally probable. Thus any design cases must cover the range of ramps that are relevant. This may be possible for single ramps by testing three cases of equi-probable shears.

a) a shear with a gradient greater than the acceleration capability of the aircraft,

b) a shear with (Speed change) about the same as the initial speed margin from stall,

c) a shear between the two above.

These shears should be chosen from the range of equi-probable shears for the normal landing and take-off speeds, and also acceleration potential of the aircraft (or class of aircraft) being considered.

The second question about San Francisco on-shore winds describes a shear of +20 kt in a ramp length of 667 m, which is normalized shear of +1.18 $(m/s)/(m^{1/3})$. Fig 22 show data for San Francisco and the on-shore winds are in the octave with a nominal ramp length of 594 m and the higher band from 250 to 500 ft. Extrapolating the main trend of the line gives a probability of 2.4/10000 per ramp length travelled for a normalized shear of +1.18 $(m/s)/(m^{1/3})$. Between 250 and 500 ft there are 2.5 shear lengths for each landing and thus this type of shear (or any of the other shears corresponding to the same normalized value) could be encountered once in 1700 landings. It should be remembered that this probability applies to landings at the normal scheduled landing time for BA B747 flights to San Francisco, which is 1545 local time.

Statistically it is not meaningful to ask for the probability of occurrence of a specific ramp but only of ramps within a band of characteristics. Thus the probability of shears with a particular normalized value can be determined, but the probability of a specific ramp is infinitesimal.

6.2 Turbulence for Design Cases

Having said that adequate representation of turbulence is important for design wind shear cases it is necessary to define some relationship between these two contributions to the total winds. One simple approach would be to have the standard deviation of the turbulence proportional to the instantaneous total wind, ie wind shear plus any general wind. The general level of fluctuations in wind speed in the individual wind shear examples of Figs 4 to 12 tends to support this suggestion in many cases. A standard deviation of horizontal turbulence of about 10% of the total wind is a reasonable level. However, there are two significant cases where this simple approach is not adequate, viz:

a) topographic turbulence, which is related to local terrain variations as well as wind velocity,

b) thunderstorm downburst where the fluctuations in the downdraught are nearly as large as the mean downdraught.

Data for Anchorage and Hong Kong suggest that topographical turbulence can have a standard deviation of about 20% of the total wind for horizontal turbulence.

The normal 'g' fluctuations of Fig 13 suggest that the standard deviation of the downdraught speed is about 50% of the mean downdraught. This figure is supported by unpublished data of a thunderstorm downburst encounter on take-off from Chicago.

Generally vertical turbulence will be less than horizontal turbulence near the ground and reduce as the height decreases. However, for many cases it may be adequate to keep vertical turbulence at about 2/3 of horizontal turbulence, which is typical for a few hundred feet above ground (see MIL-F-8785C, Fig 11). A decrease in vertical turbulence with height for thunderstorm downdraught cases will be generated automatically as the downdraught decreases near the ground.

Generating the progression of turbulence in all three axes for a path through a thunderstorm downburst requires knowledge of the location of the downburst. This is best identified by the presence of vertical wind (updraught or downdraught). It is then possible to set turbulence as a basic level depending on horizontal wind which applies both within and outside the downburst, plus additional turbulence related to the downdraught (or updraught). This will give the very high levels of turbulence experienced in downbursts.

Table 5 summarises the suggested levels of turbulence for various shear cases. No attempt has been made to simulate the more subtle variations of turbulence with height, wind direction, temperature gradient, etc. The aim is to provide turbulence that is sufficiently representative of actual operations to test the effectiveness of systems that react to or detect wind shear. More subtle descriptions of turbulence are probably unnecessary, although if they are already available in a simulation then they can be used.

It is, however, important that the turbulence simulation has the intermittency and distribution of gusts found in the real atmosphere. This is best achieved using the Discrete Gust model of Ref 8 which has been validated in normal simulations.

7 FURTHER WORK

The data from the BA B747 landings are sufficient to define suitable realistic design cases for wind shear. However, although the worldwide coverage is extensive, it is very desirable to obtain further data from other sources to confirm the levels of probability and obtain detailed information on other airports and/or landing times. To do this the data must be analysed by the Discrete Gust method. The Dutch National Aeronautical Laboratory (NLR) are planning to use the Discrete Gust methods on about 8000 landings already stored on their computer (Ref 9). When this is completed then the RAE and NLR will exchange data. Any data from other sources can be added to this data base if it is analysed by the same method.

Another essential activity to validate design cases is to continue analysing large wind shear incidents and this will be done by including large wind shears in the UK Civil Aircraft Airworthiness Data Recording Programme (CAADRP) together with information on the total number of landings searched. This same data will also provide a basis for confirming severity scales by relating the severity of the shear to the aircraft height loss and pilots' reaction. Again, it would be extremely valuable if other agencies and/or airlines could provide data analysed in the same way.

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A) RECORDED DATA

	Parameter	Resolution	Samples/sec	Comments
Run Data	Airport	-	-	
	Runway Heading	1 deg	-	Magnetic
	Touchdown Weight	0.1 tonne	-	
Recorded Data	Groundspeed	0.1 kt	1	
	Calibrated Air Speed	0.1 kt	1	
	Pressure Altitude	1 ft	1	
	Total Air Temperature	0.1 deg C	0.5	
	Windspeed	1 kt	0.25	)INS smoothed
	Wind Direction	1 deg	0.25	)outputs used)for verification)purposes
	Pitch Attitude	0.1 deg	4	
	Roll Attitude	0.1 deg	2	
	Heading	0.1 deg	1	Magnetic
	Normal Acceleration	0.001 g	8	
	Lateral Acceleration	0.001 g	4	
	Longitudinal Acceleration	0.001 g	1	
	No 2 Engine Pressure Ratio	0.001	1	
	No 3 Engine Pressure Ratio	0.001	1	Spare
	ILS Glide Slope	0.1 mV	1	
	ILS Localizer 1	0.1 mV	1	
	ILS Localizer 2	0.1 mV	1	
	ILS Localizer 3	0.1 mV	1	
	Flap Angle	1 deg	1	
Discrete Data	Autopilot A Engaged	-	1	
	Autopilot B Engaged	-	1	
	Autopilot C Engaged	-	1	
	Localizer 1 Captured	-	1	
	Localizer 2 Captured	-	1	
	Localizer 3 Captured	-	1	
	Autothrottle Engaged	-	1	
	ILS Deviation Warning	-	1	

NB: Data is provided from 125 seconds before touchdown to 10 seconds after it.

B) CALCULATED DATA

All results at one per second from 123 seconds before touchdown through to touchdown.

Wind: Horizontal speed (kt)
Horizontal direction from runway (deg)
Vertical speed (ft/min)

Static Air Temperature (deg C)
Pressure height above touchdown (ft)
Pressure change at constant height (mb)
Inertial vertical velocity (ft/s)
Sideslip angle (deg)
Incidence angle (deg) (NB: No incidence record available)
Drift angle (deg)
Engine pressure ratio (No 2)

TABLE 1. ALERT DATA INPUTS AND OUTPUTS (WSALE)

Airport		Total Visits	Approx Scheduled BA B747 Landings(March 1982)	
Code	Name		Weekly	Local Time (No per Week)
LHR	London(Heathrow)	2413	C.120	All day
NBO	Nairobi	602	14	0800(7),2330(7)
BOM	Bombay	447	22	0030(3),0230(5),0435(2),0615(3),0900(2), 1305(1),1510(1),1615(1),2230(1),2355(1)
SYD	Sydney	281	20	0615(6),0740(7),1555(2),1600(5)
SIN	Singapore	266	18	1505(1),1735(7),1820(4),2110(4),2250(1), 2340(1)
JFK	New York(JFK)	244	7	1335(7)
AUH	Abu Dhabi	243	12	0110(1),0125(2),0205(1),0400(2),0440(1), 0455(2),0645(1),2330(2)
JNB	Johannesburg	242	16	0735(1),0805(2),0840(1),1005(4),1020(1), 1150(7)
BOS	Boston	220	7	1340(3),1420(4)
BAH	Bahrain	212	9	0050(2),0220(1),0250(1),0315(2),0330(1), 0620(1),2300(1)
SFO	San Francisco	211	7	1545(7)
KUL	Kuala Lumpur	187	8	0025(1),1620(2),1650(2),2325(3)
SEA	Seattle	184	3	1255(3)
PER	Perth	182	8	0020(2),1840(1),2130(2),2325(3)
MEL	Melbourne	181	16	0700(2),0900(4),1000(1),1020(7),1920(2)
IAD	Washington DC(IAD)	178	7	1445(4),1615(3)
LAX	Los Angeles	171	5	1525(5)
FCO	Rome (FCO)	167	4	0525(1),0605(1),1250(1),1920(1)
MCT	Muscat	165	7	0320(1),0410(2),0430(1),0520(1),0615(1), 0710(1)
YYZ	Toronto	157	9	1545(2),1610(1),1730(4),1735(2)
NRT	Tokyo (NRT)	146	12	1450(4),1545(2),1720(2),1930(1),2030(3)
BGI	Barbados	139	7	1540(1),1725(1),1735(2),2045(1),2230(2)
HKG	Hong Kong	137	12	1015(1),1110(1),1115(2),1505(3),1640(1), 1945(3),2045(1)
ANC	Anchorage	127	8	0855(4),1110(4)
YMX	Montreal	125	6	1340(2),1535(4)
MIA	Miami	103	7	1730(7)
PIK	Prestwick	97	1	0710(1)
SAY	Harare	91	2	0540(2)
ORD	Chicago (O'Hare)	81	5	1645(5)
DTW	Detroit	74	4	1635(4)
PHL	Philadelphia	72	3	1630(3)
AKL	Aukland	66	7	0735(1),0745(4),1315(2)
CMB	Colombo	65	4	0850(1),1635(1),1815(1),2115(1)
MAN	Manchester	63	1	0720(1)
OSA	Osaka	61	7	1645(1),1705(3),2015(3)
CCU	Calcutta	59	2	0950(1),2235(1)
POS	Port of Spain	58	3	1735(1),1930(2)
MNL	Manila	57	2	1420(2)
ZRH	Zurich	42	2	0535(1),1430(1)
SEZ	Seychelles	40	4	0340(1),0545(1),1910(1),1945(1)
BKK	Bangkok	37	4	0010(1),1430(1),1435(1),1440(1)
HLP	Jakarta	36	1	1055(1)
LGW	London (Gatwick)	34	0	
DXB	Dubai	33	0	
BWN	Brunei	32	2	1855(1),2110(1)

TABLE 2. SUMMARY OF LANDINGS AND SCHEDULED TIMES

Airport		Total Visits	Approx Scheduled BA B747 Landings(March 1982)	
Code	Name		Weekly	Local Time (No per Week)
STN	Stansted	31	0	
PEK	Peking	30	2	1355(1),1405(1)
DUR	Durban	28	2	0855(1),1305(1)
ANU	Antigua	26	4	1525(1),1620(1),2115(1),2155(1)
YVR	Vancouver	26	2	1710(1),1730(1)
BNE	Brisbane	25	2	0800(2)
DHA	Dhahran	24	1	2155(1)
YYC	Calgary	22	1	1620(1)
SNN	Shannon	21	0	
UVF	Saint Lucia	16	1	1815(1)
MRU	Mauritius	15	1	0915(1)
YEG	Edmonton	11	1	1600(1)
KWI	Kuwait	5	4	1920(2),2100(1),2125(1)
FRA	Frankfurt	4	0	
BHX	Birmingham	3	0	
JED	Jeddah	3	0	
CDG	Paris (CDG)	3	0	
ATH	Athens	3	0	
DEL	Delhi	3	10	0330(2),1115(1),1220(1),1325(1),2340(5)
EDI	Edinburgh	2	0	
AMM	Amman	2	0	
BRU	Brussels	1	0	
PMI	Palma	1	0	
LCA	Larnaca	1	0	
MBA	Mombasa	1	0	
LPL	Liverpool	1	0	
	Total	9136		

TABLE 2 (Contd)

Intensity of Shear	Effect on Aircraft Control	Head Wind Shear in 600 m*
Light	Little	0 to 2 m/s (0 to 3.9 kn)
Moderate	Significant	2 to 4 m/s (4 to 7.8 kn)
Strong	Considerable difficulty	4 to 6 m/s (7.9 to 11.7 kn)
Severe	Hazardous	>6 m/s (11.7 kn)

*Described separately as change for 600 m horizontally and 30 m in height, but this is equivalent to 600 m travelled on a 3° flightpath

TABLE 3. WINDSHEAR SEVERITY SCALE PROPOSALS TO ICAO

Nominal Ramp Length, m	Ramp Lengths for 1000 Landings	$\frac{\text{(Speed change)}}{\text{(Ramp length)}^{1/3}}$ (m/s)/m $^{1/3}$	Ramp Length m	Speed Change kt	Gradient kt/100 m
305	30000	-1.04	200 300 400	-11.8 -13.5 -14.9	-5.9 -4.5 -3.7
609	15000	-0.85	400 600 800	-12.2 -13.9 -15.3	-3.0 -2.3 -1.9
1218	7500	-0.78	800 1200 1600	-14.1 -16.1 -17.7	-1.8 -1.3 -1.1

TABLE 4. SINGLE RAMPS WIND SHEARS WITH A 1 in 1000 LANDINGS PROBABILITY. WORLDWIDE DATA (FIG 16)

Case	Standard Deviation of		
	Headwind u	Crosswind v	Downdraught w
General Wind Shear	0.1V	0.1V	0.07V
Topographical Wind Shear	0.2V	0.2V	0.13V
Thunderstorm Downburst Wind Shear	(0.1V + 0.2D)	(0.1V + 0.2D)	(0.07V + 0.4D)

V = Horizontal Total Wind before adding turbulence
D = Downdraught in thunderstorm downburst before adding turbulence

NB: Both these are treated as positive regardless of wind direction.

TABLE 5. SUGGESTED TURBULENCE LEVELS FOR WIND SHEAR DESIGN CASES

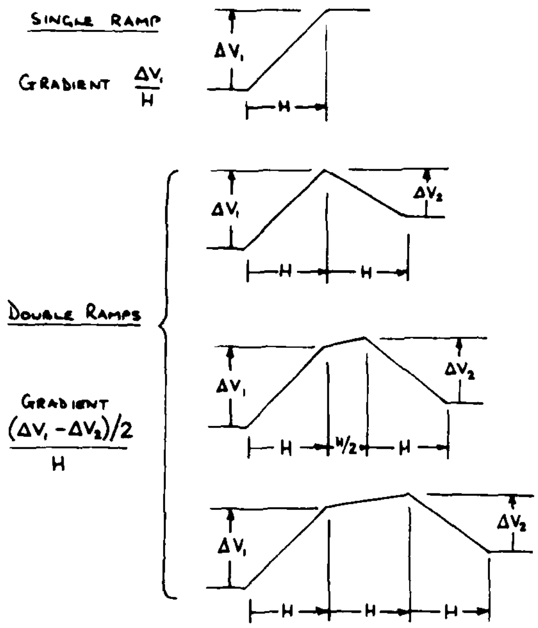
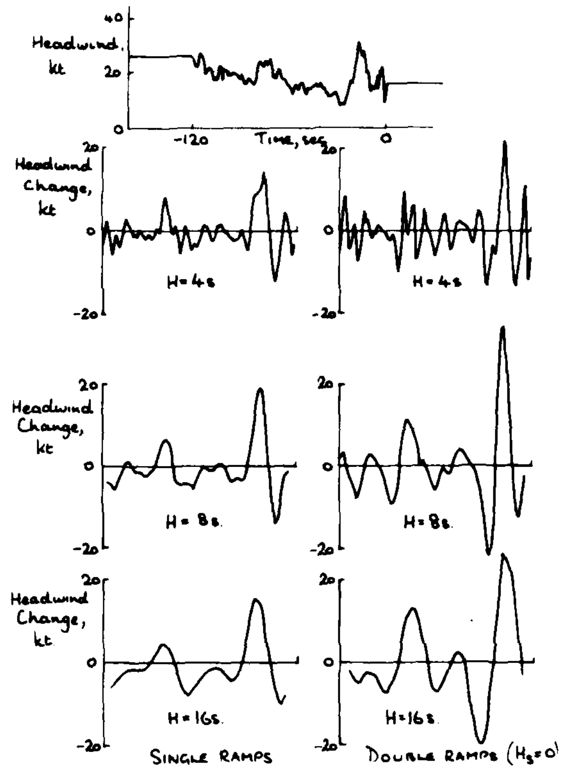


Fig. 1 Windshear patterns

a. Initial time history



b. Filtered time histories

Fig. 2 Windshear pattern detection Landing at San Francisco, April 1982

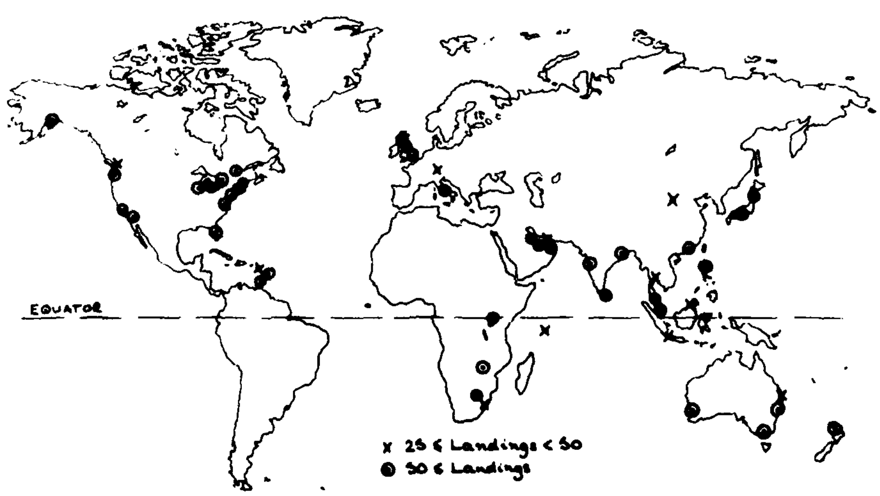
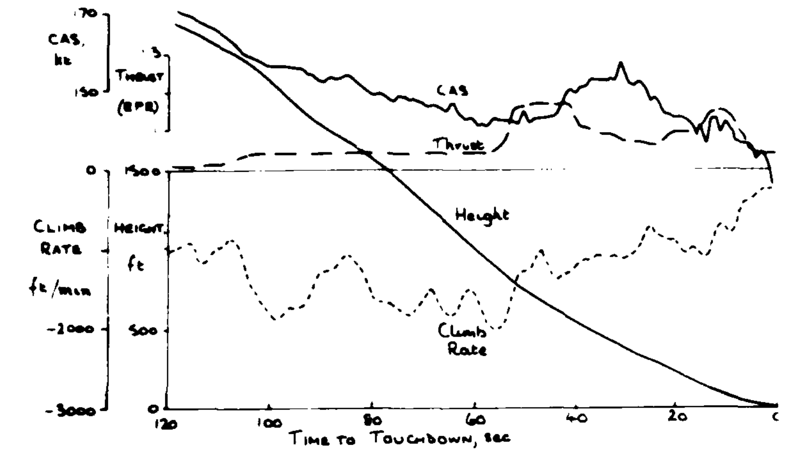
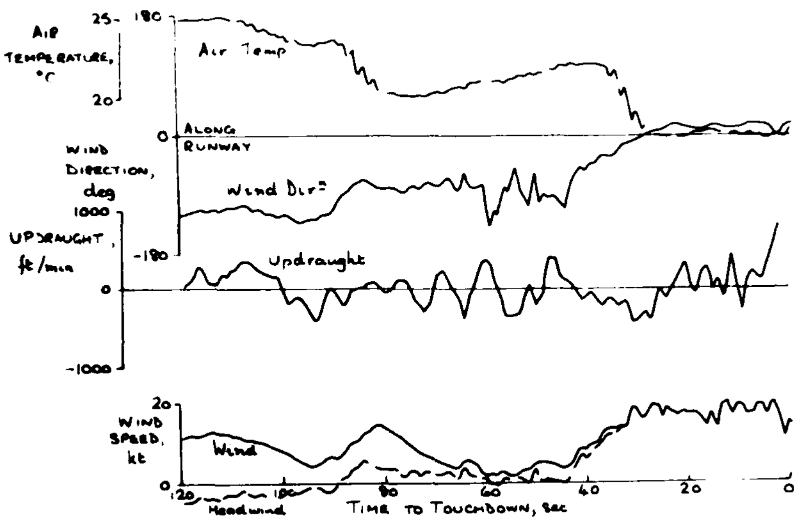


Fig. 3 Distribution of airports visited by British Airways B747 aircraft

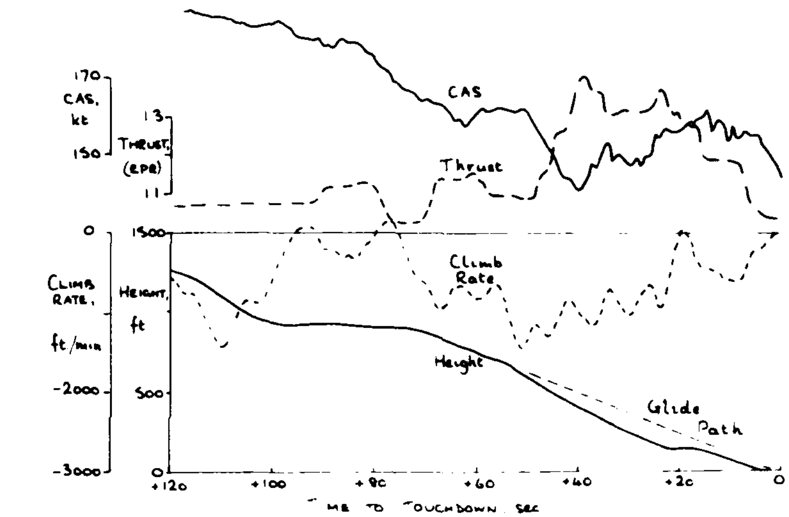


a. Aircraft reaction

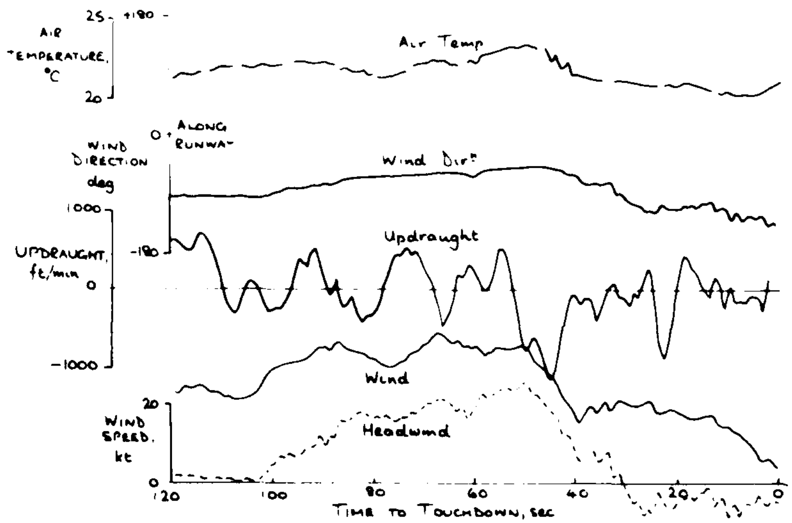


b. Winds and temperature

Fig. 4 Windshear at San Francisco, August 1981

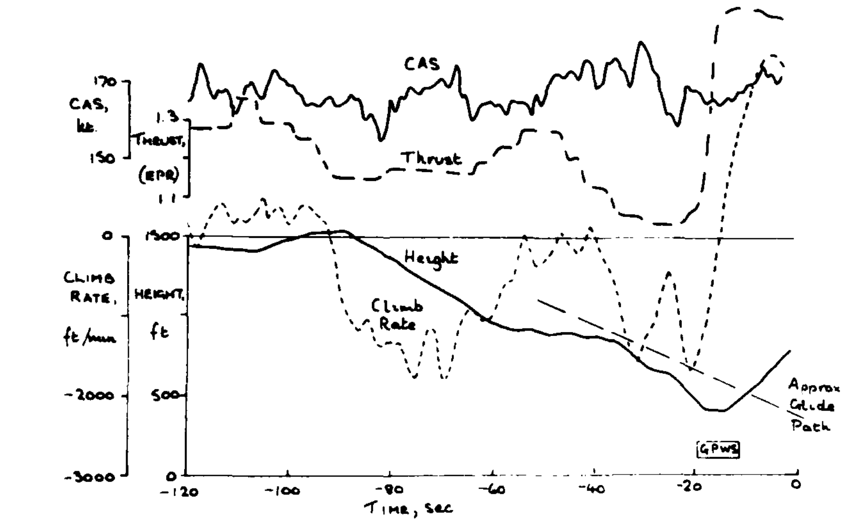


a. Aircraft reaction

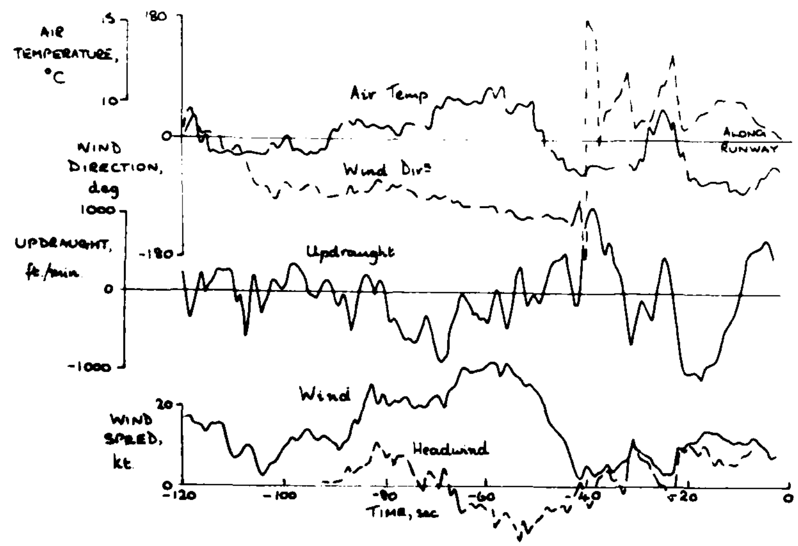


b. Winds and temperature

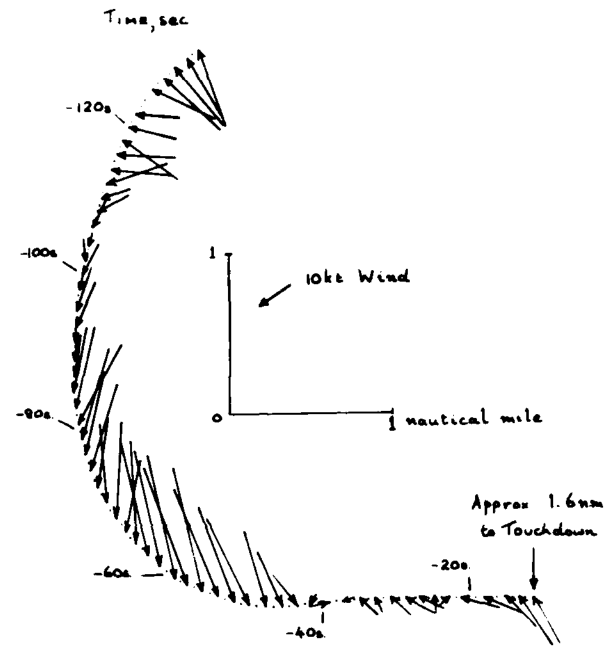
Fig. 5 Windshear at Melbourne, September 1961



a. Aircraft reaction

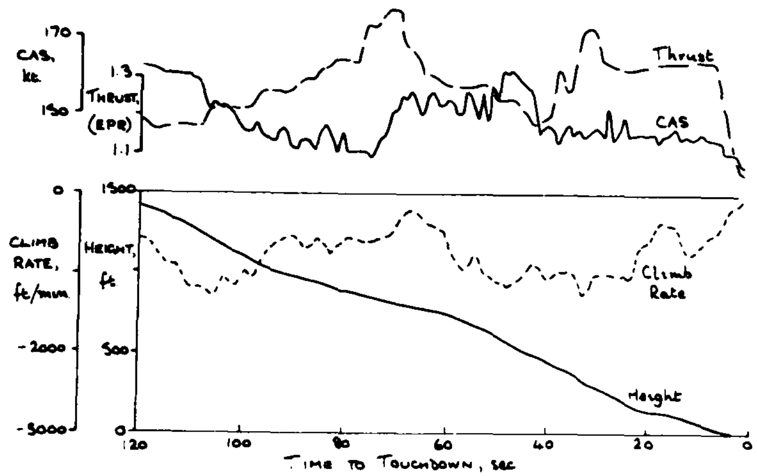


b. Winds and temperature

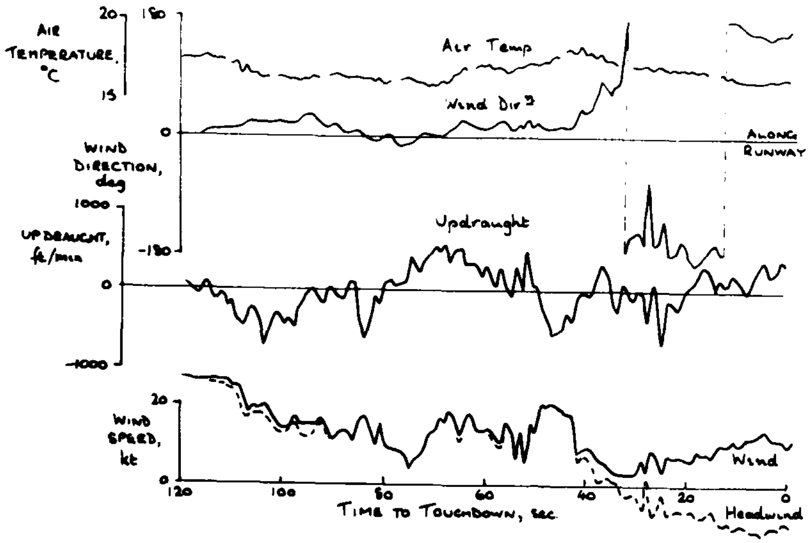


c. Ground track and wind vectors

Fig. 6 Windshear at Anchorage, March 1981

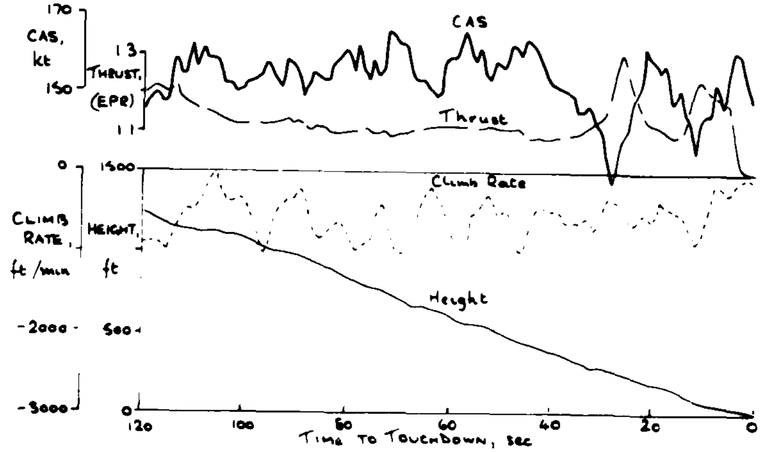


a. Aircraft reaction

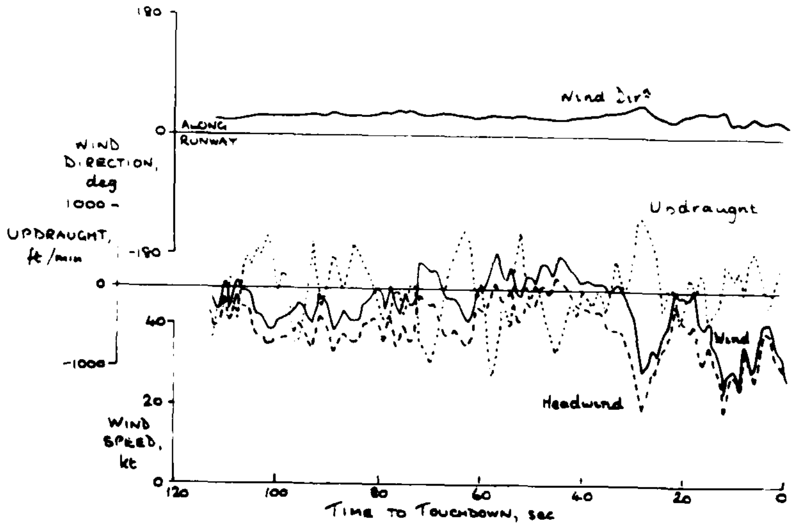


b. Winds and temperatures

Fig. 7 Windshear at Harare, Zimbabwe, August 1981

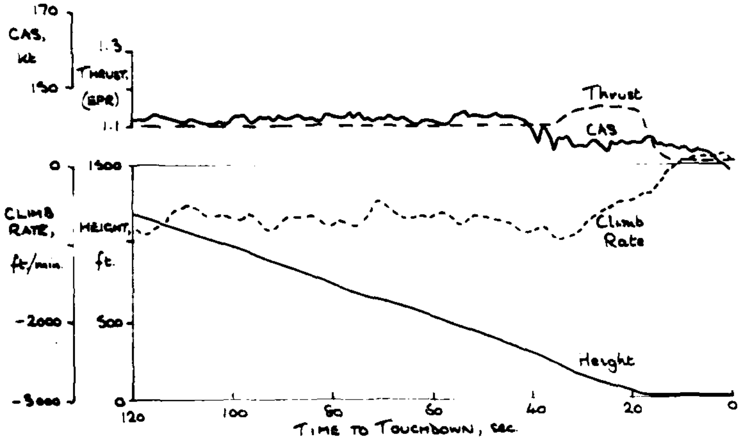


a. Aircraft reaction

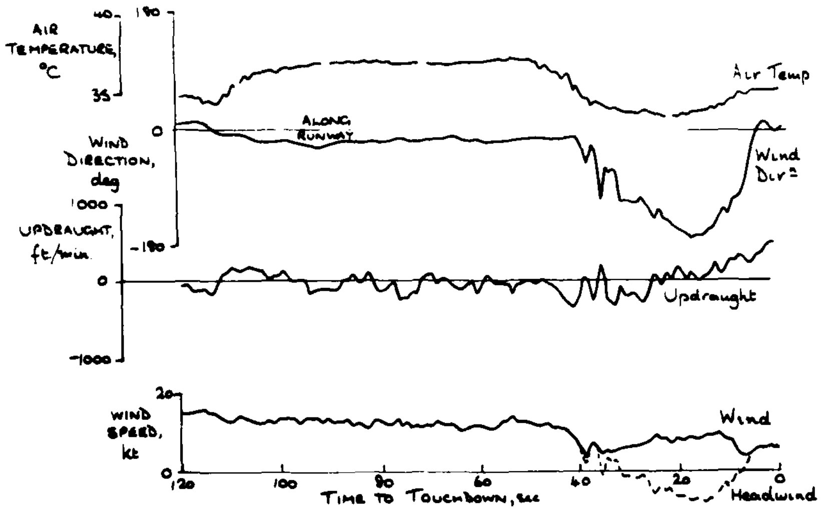


b. Winds

Fig. 8 Windshear at Detroit, April 1982

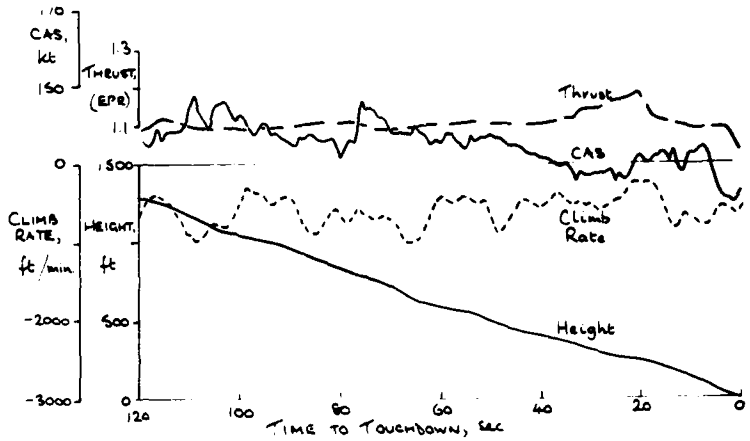


a. **Aircraft reaction**

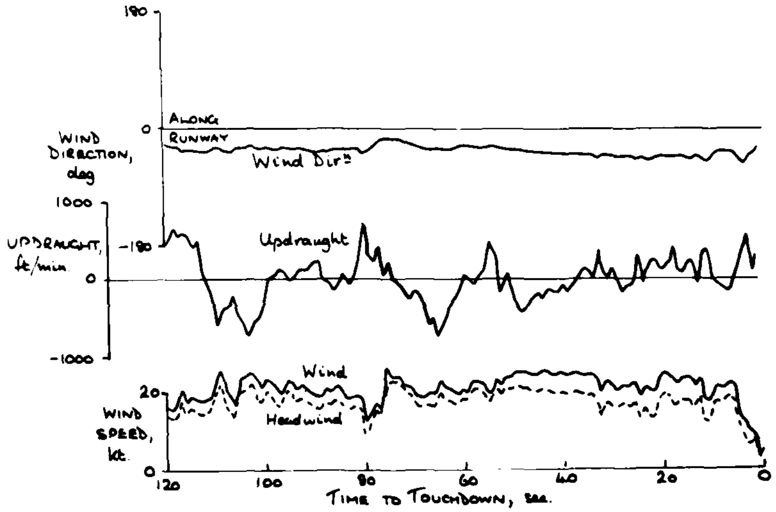


b. **Winds and temperature**

Fig. 9 **Windshear at Muscat, June 1982**

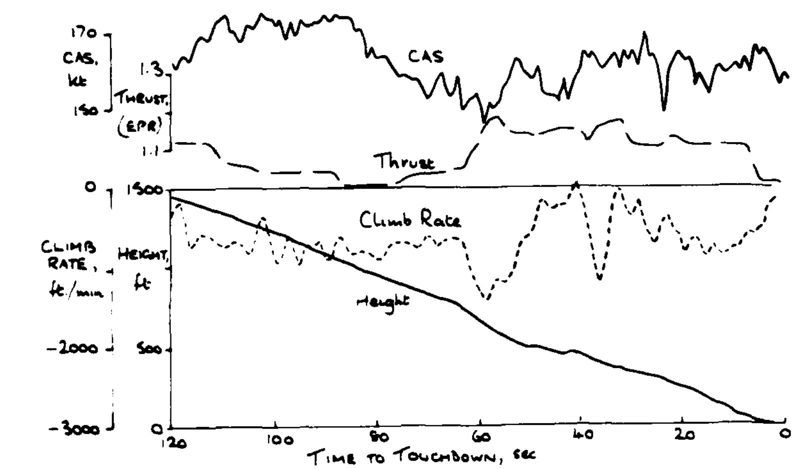


a. **Aircraft reaction**

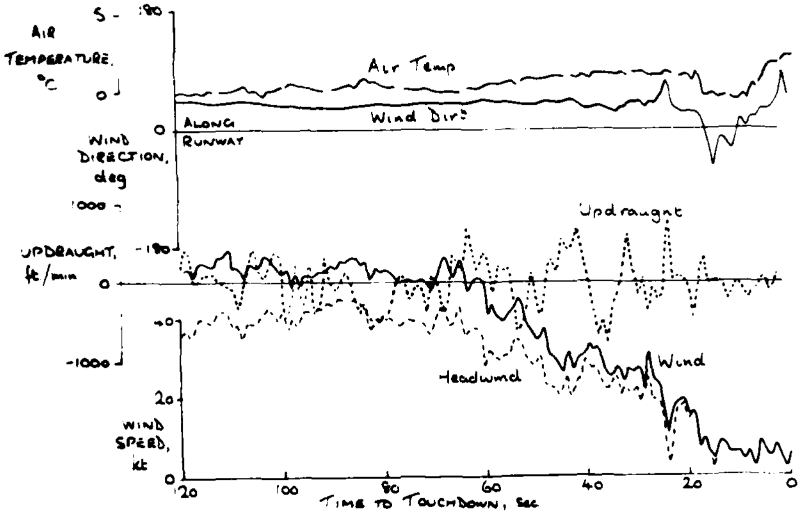


b. **Winds**

Fig. 10 **Windshear at Chicago, April 1982**

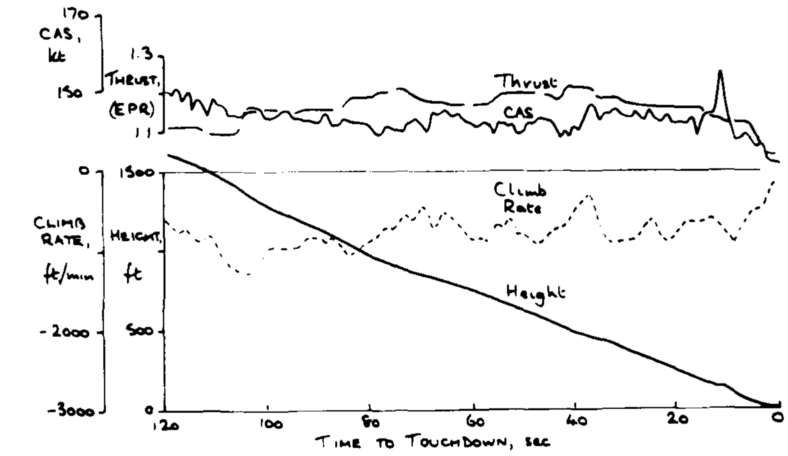


a. Aircraft reaction

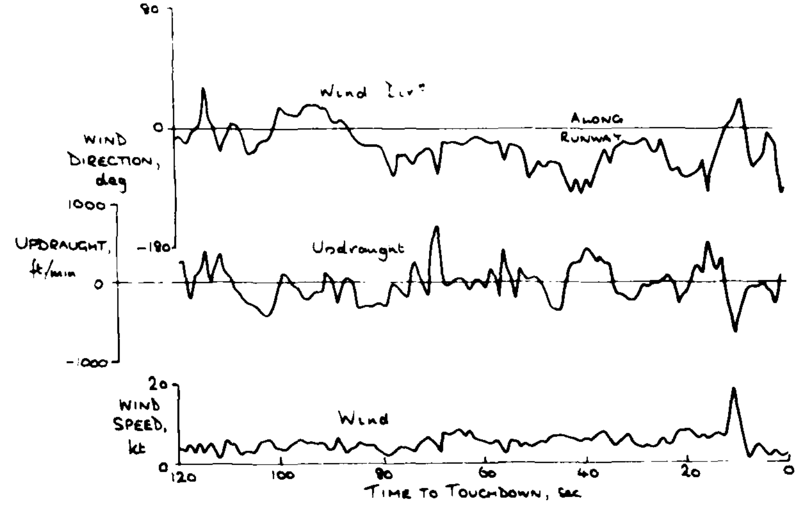


b. Winds and temperature

Fig. 11 Windshear at Anchorage, November 1981



a. Aircraft reaction



b. Winds

Fig. 12 Windshear at London (Heathrow), April 1982

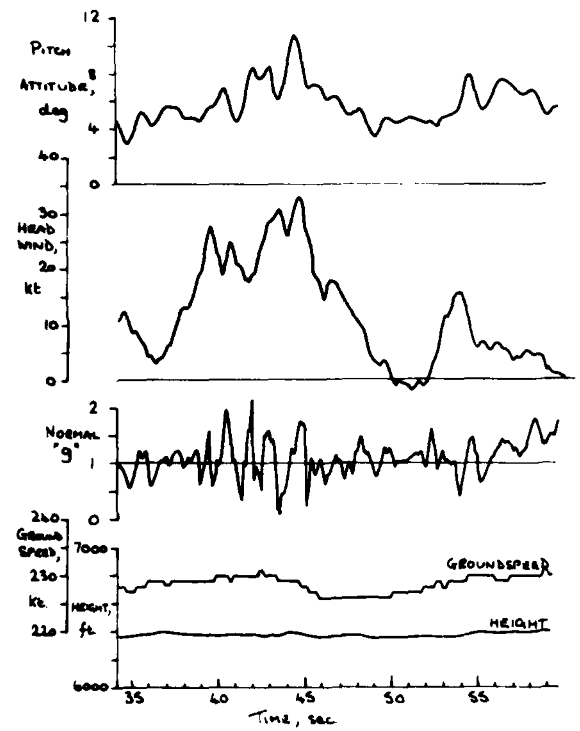


Fig. 13 Thunderstorm microburst - JAWS Project, RAE HS125 - Flight 792, Run 3

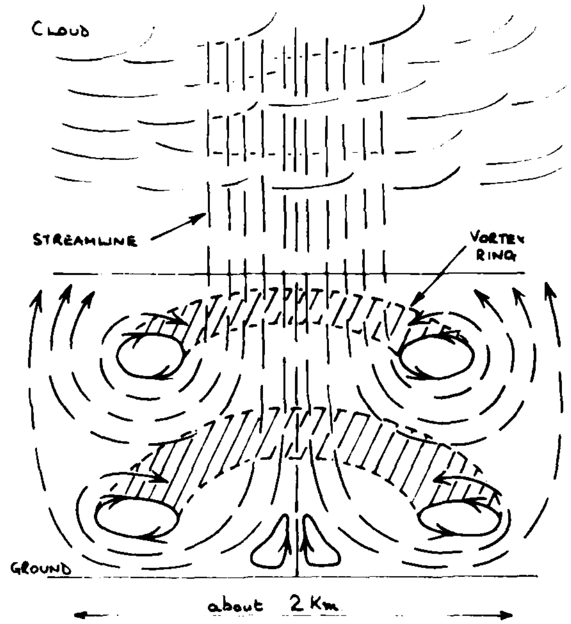


Fig. 14 Thunderstorm downburst. Vortex ring flow model

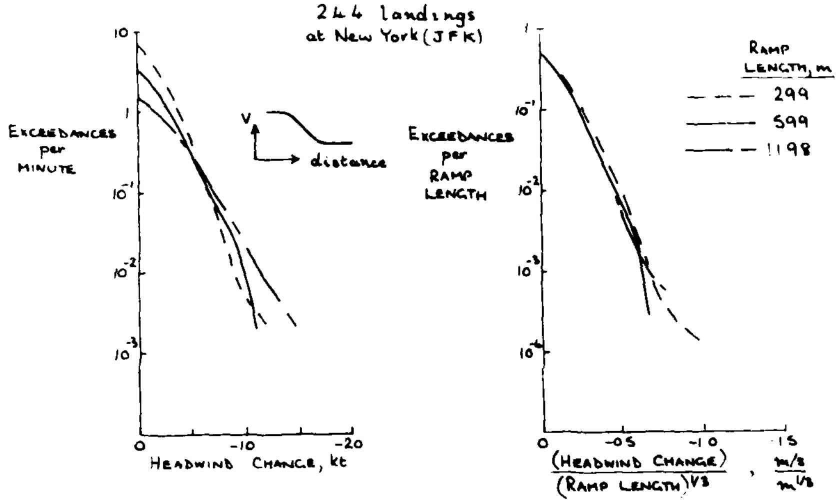


Fig. 15 Normalizing windshear by ramp length. Single ramps

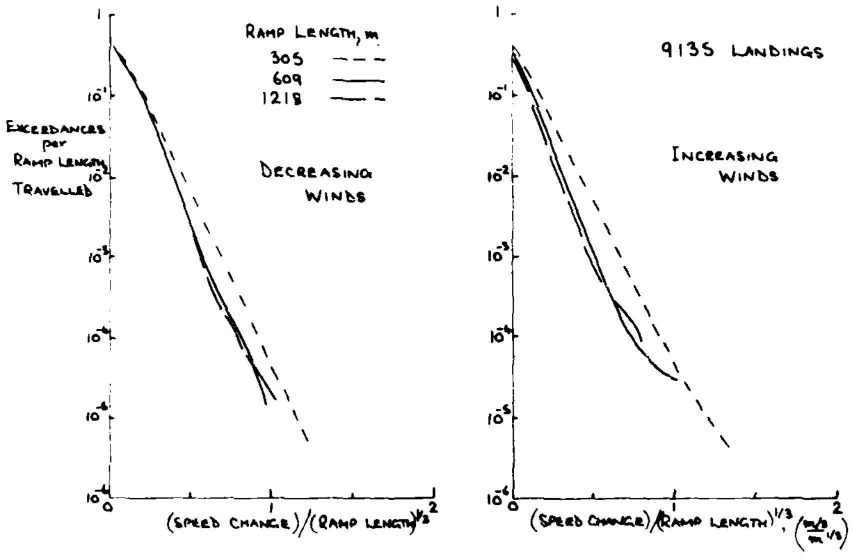


Fig. 16 Cumulative distribution of single ramps (British Airways records)

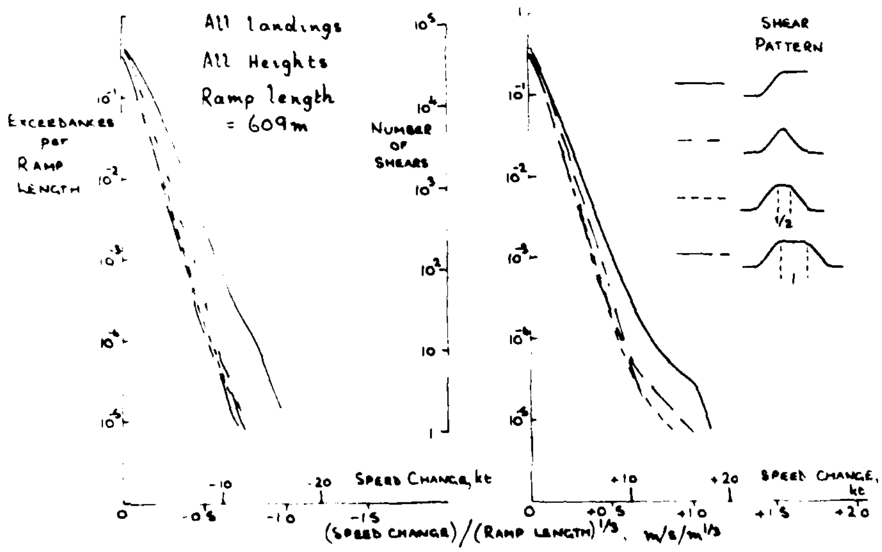


Fig. 17 Variation of shear probability with pattern

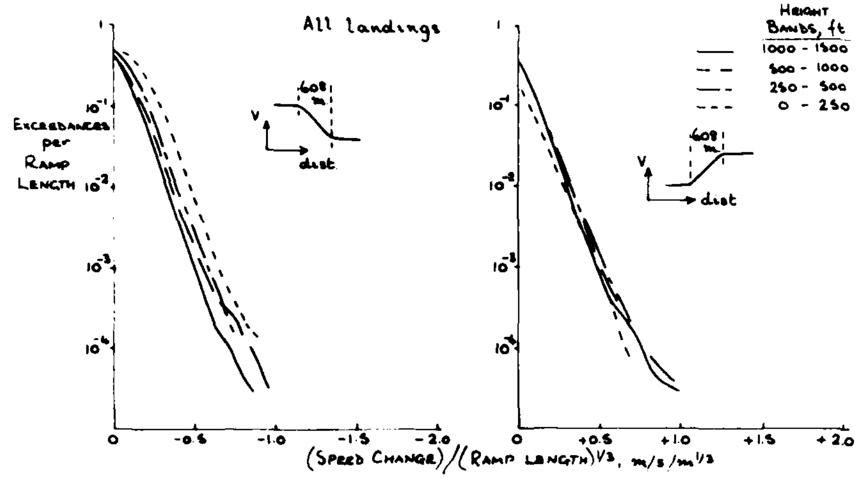


Fig. 18 All landings. Variation of 608 m single ramps with height

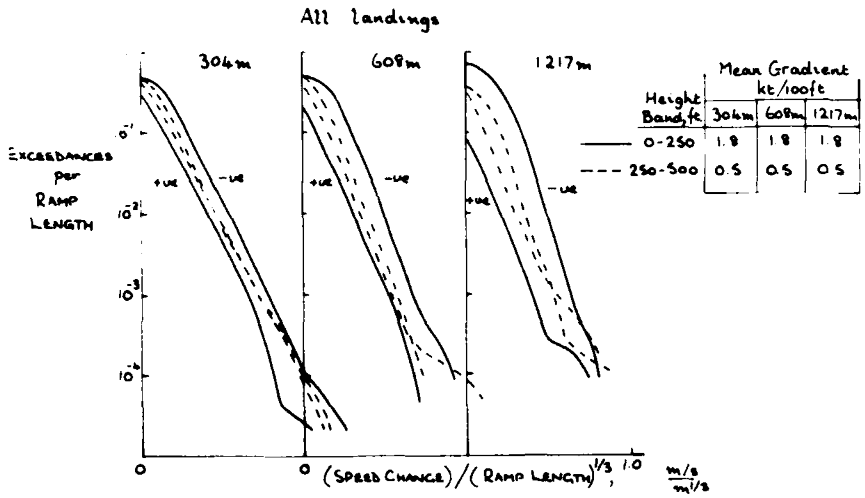


Fig. 19 Effect of mean gradient on exceedances. Single ramps

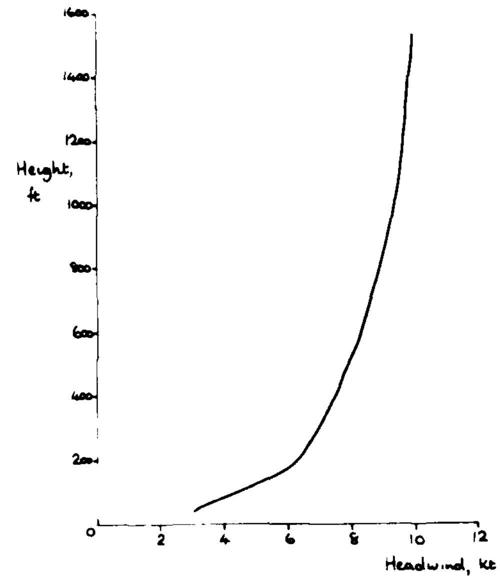


Fig. 20 Mean headwind variation with height

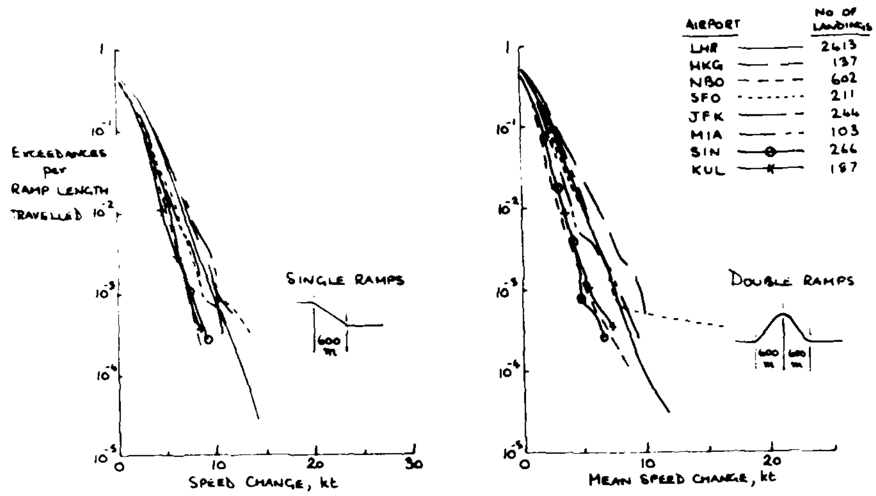


Fig. 21 Cumulative distributions of single and double 600 m ramps at a selection of airports

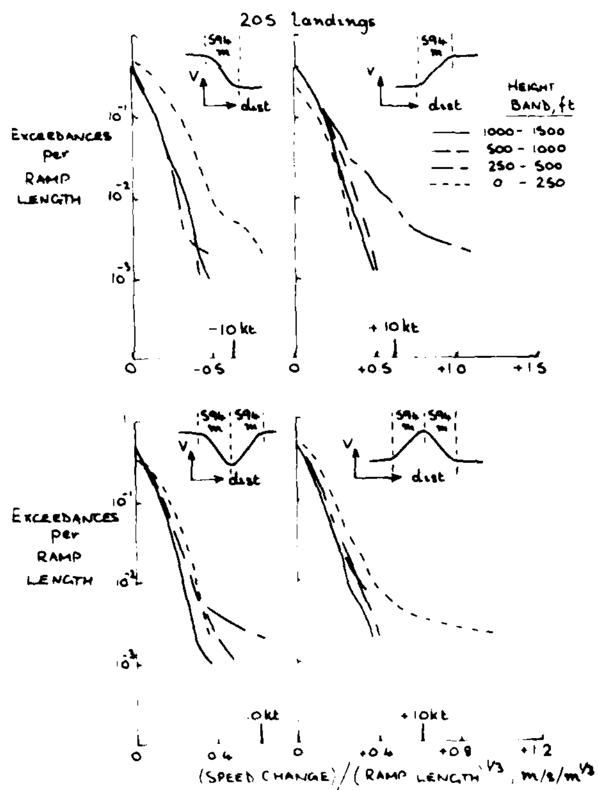
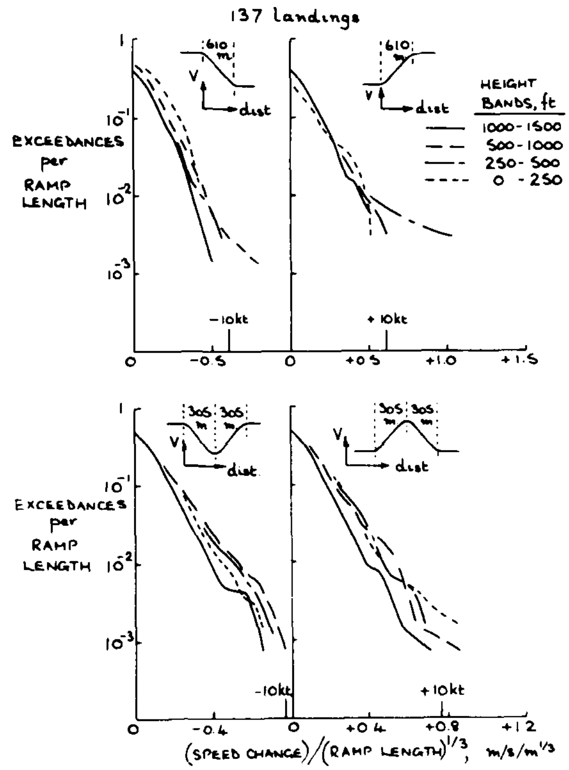
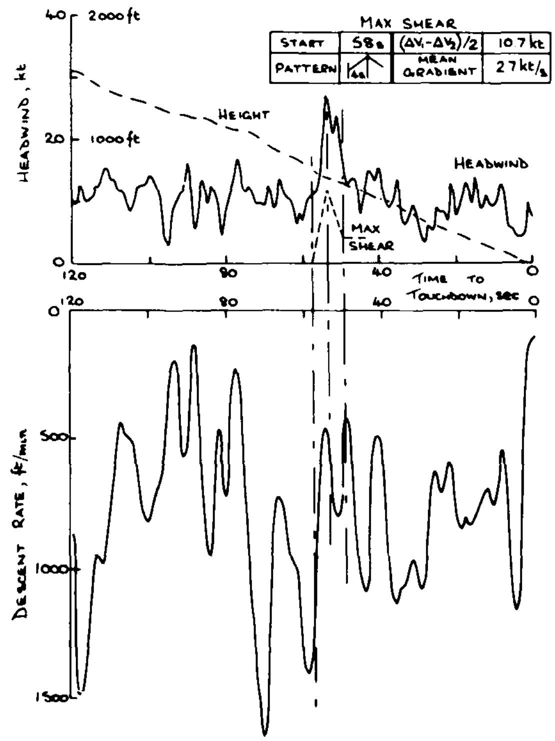


Fig. 22 San Francisco. Variation of single and double ramps with height



a. Variation of single and double ramps with height



b. Mountain wake windshear (Ref 4, Fig 6)

Fig. 23 Hong Kong (R/W 31)

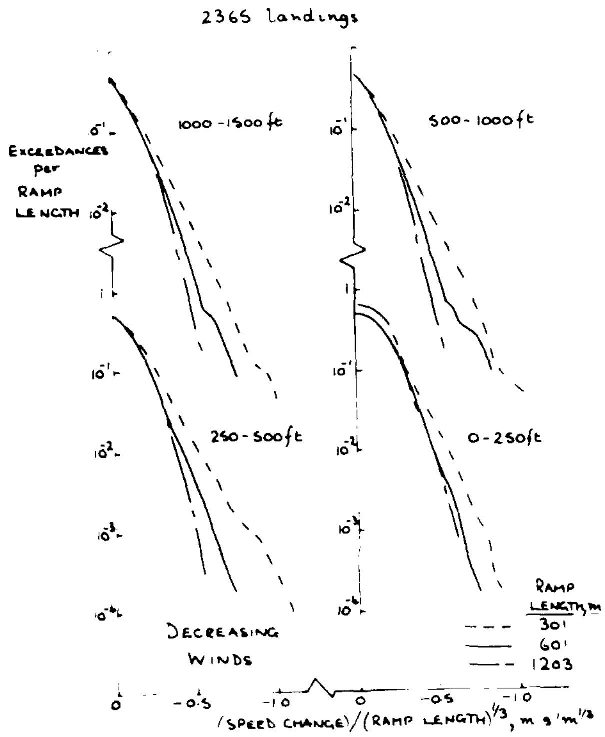


Fig. 24 London (Heathrow). Variation of single ramps with height

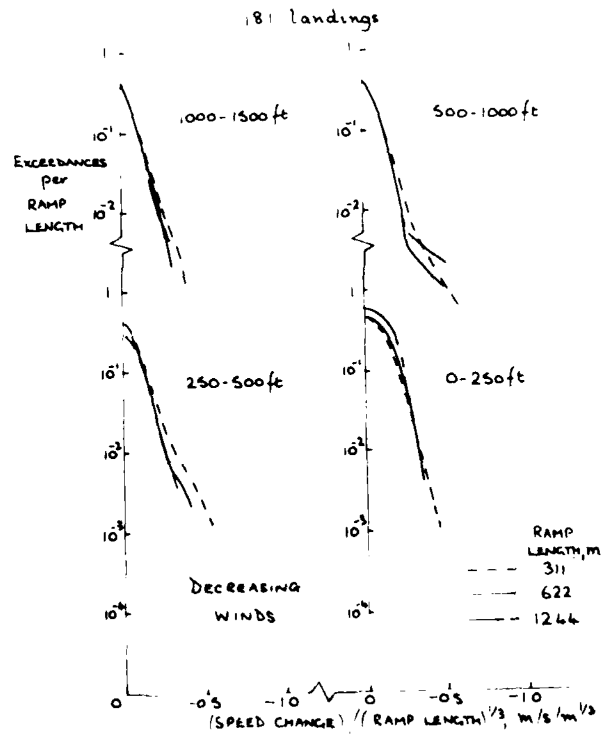
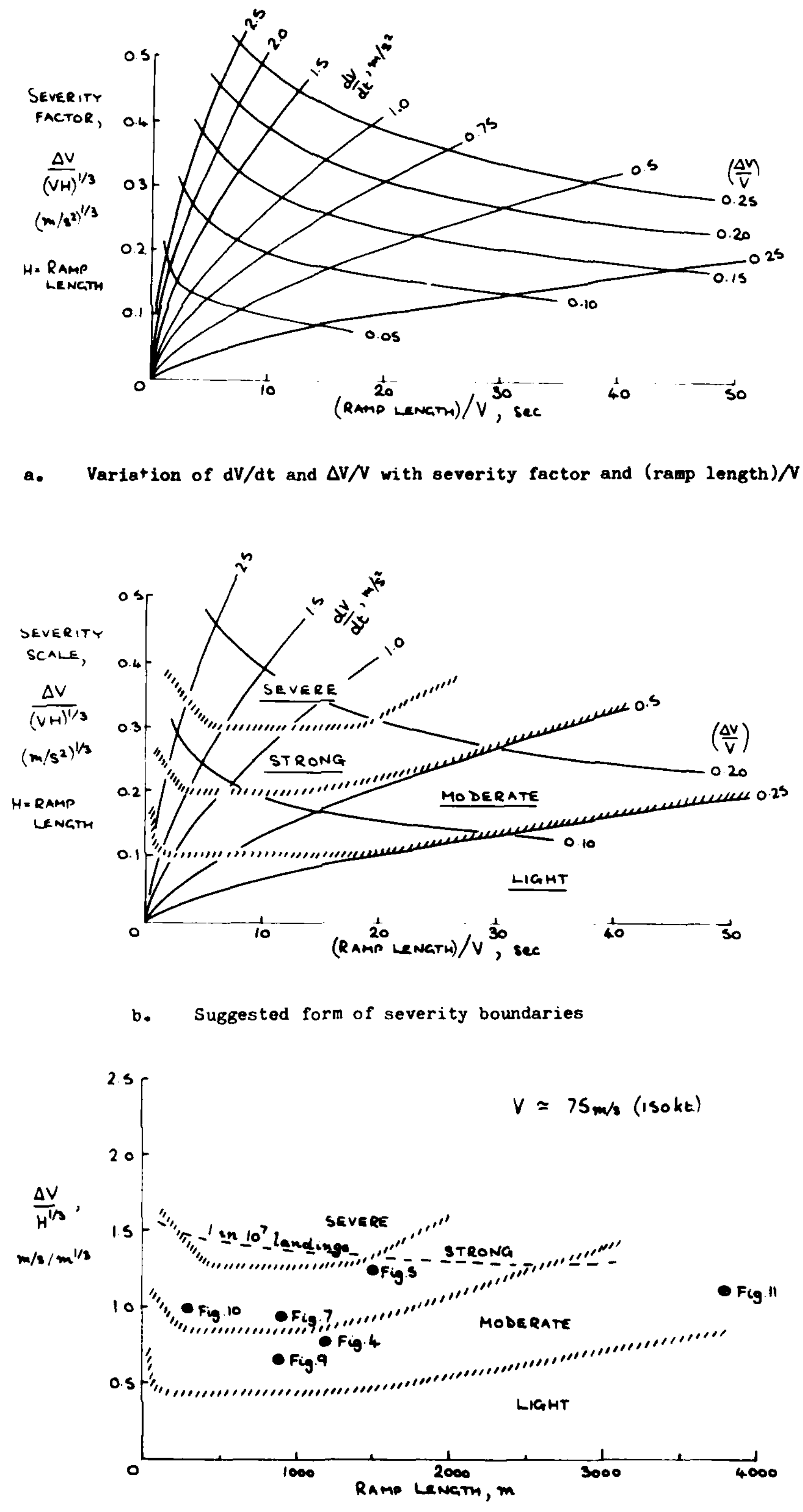


Fig. 25 Kuala Lumpur. Variation of single ramps with height

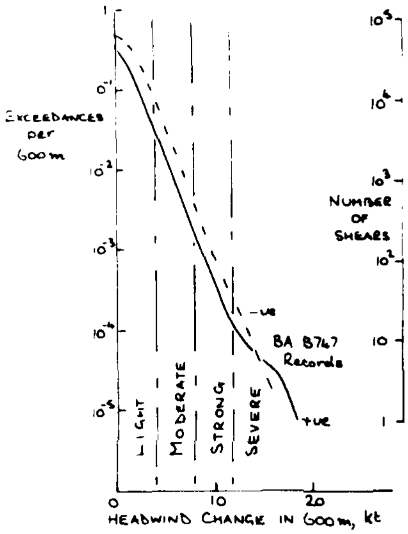


a. Variation of dV/dt and ΔV/V with severity factor and (ramp length)/V

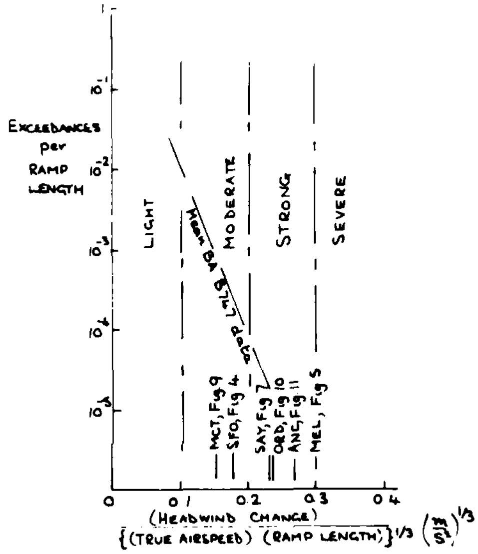
b. Suggested form of severity boundaries

c. Severity scales for B747

Fig. 26 Possible severity scales for single ramp headwind changes



a. Severity scale suggested at ICAO



b. Possible simplified severity scale

Fig. 27 Simplified severity scales

INFLUENCE OF WINDSHEAR ON FLIGHT SAFETY

by

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1 INTRODUCTION

Wind shear during take-off and landing may crucially restrict flight safety. In some rare situations, especially during take-off, hazards may be caused by limited flight performance.

In most cases wind shear accidents and incidents result from the fact that the wind shear phenomenon is not understood by the pilot due to his training condition and the cockpit instrumentation. In such situations the pilot is not able to act in the correct way. Therefore it can be suspected that a considerable amount of wind shear accidents will be interpreted wrongly as pilot's error.

Numerous investigations have been made in order to solve the wind shear problem. Many of these proposals will fail because the physical phenomena are not understood completely, neither by the pilots nor by the investigators of the wind shear warning system. This problem will be illuminated by the fact that some of the correct safety procedures in wind shear contradict the pilot's feeling of how to control an aircraft.

This paper tries to clarify step by step some physical backgrounds of the wind shear phenomena including adequate flight safety procedures to overcome the problems.

2. WIND SHEAR SCENARIO

Wind shear is an alteration of horizontal wind with time, height and distance. Concerning the airplane this results in a time varying wind speed.

Wind shear is not only related to stormy weather, as one might expect, but also to misty mornings in early summer as well as to bright sunshine in periods of fair weather [1]. In most cases wind shear is combined with up- and downdrafts.

Wind shear as a great variety of phenomena may be characterized by two extreme situations:

- thunderstorm with immense time varying effects, high turbulence and extreme downdrafts; (fig.1a)
- low level jet lasting for hours, no turbulence, less wind velocity outside the jet (fig.1b).

Between these two extremes several other kinds of wind shear related weather phenomena have been observed.

In addition flight safety may be reduced by wake turbulence caused by large buildings as well as by orographic lee effects. The effect of surface boundary layer is always persistent [2], (fig.2). Wake turbulence caused by large buildings (fig.3) can reduce flight safety as well as orographic lee effects (fig.4)

Due to geographic circumstances and the thickness of the shear layer, small or moderate wind shear gradients can be more dangerous in some exceptional situations than extreme thunderstorm shears [3].

3. AIRCRAFT RESPONSE DUE TO WIND SHEAR

.e time varying wind changes first the airspeed of an aircraft due to its inertia. As a consequence varying aerodynamic forces accelerate the aircraft. Due to the static stability of the airplane and due to the pilot's behaviour to keep the airspeed constant, the aircraft will be accelerated by the varying wind. The airspeed deviation is amazingly small (fig.5). The largest airspeed deviation is caused by the dynamic response of the aircraft outside the shear layer 1]. The more important flight path deviations are demonstrated in fig.7 for take-off and in fig.6 for landing approach.

Investigations on energy transfer between wind and aircraft [1,4] have shown that energy based on inertial speed and height is fairly constant at small flight path angles. This is true for all transport aircraft in normal flight regimes (fig.5). Only glider aircraft can transfer significant energy from the wind in extreme flight manoeuvers, as for example in dynamic soaring flight. P.Krauspe 5] pointed out that wind shear induced by flight path deviations can be approximated by simple analytical functions (fig.8). A fundamental result of this investigation was that the aircraft response in wind shear is to a great extent independent of aircraft characteristics. The major parameters of influence are airspeed and lift to drag ratio. It should be noted that the earth-fixed wind shear can extensively modify the phugoid stability (fig.9). Krauspe's numerical calculations have been verified in a moving cockpit simulation. Fig.10, curve 1 shows the response of a wide body aircraft flown by an experienced airline pilot in a reported thunderstorm situation that caused the crash of a Boeing 727 in New York in 1975. Fig.10 delivers two essential answers:

- airspeed deviation is negligibly small even under adverse weather conditions (<10 kts);
- the pilot does not respond. The flight path is very similar to the fixed control situation in fig.8.

Fig.11 shows the response of the same aircraft in the same thunderstorm and downburst situation, when the aircraft is controlled by a less experienced airline pilot (1500 flight hours). The high control activity is typical but does not prevent the crash. The airspeed deviation is very low, too.

TAKE-OFF

The influence of wind shear on flight safety during take-off and go around differs very much from the situation during approach and landing. Handling qualities, cockpit instrumentation and training condition of the pilot are essential during landing approach. On the other hand limited flight performance, especially in a "one engine- out" situation of a twin-engined aircraft dominates during take-off, go around and missed approach. The typical response of an aircraft during take-off in a wind shear is illustrated in fig.7. Primarily dangerous is the strong reduction of flight path angle. The crash of a Continental Airlines Boeing 727 in Denver, Colorado in August 1975 in a thunderstorm is shown in fig.12 3,9].Under the condition of a decreasing headwind that changed to an increasing tailwind coinciding with a strong downdraft, the aircraft struck the ground approximately 1 nm after lift off. This has been an accident that was unavoidable due to the severity of the wind shear encounter and the aircraft performing near its maximum capability.

ÓROGRAPHIC INFLUENCED WIND SHEAR

Less-known and understood is the potential hazard posed by low-velocity downdrafts and wind shear in the lee of a large-surface obstacle (fig.4). The influence of these factors has been studied at a German airport and with extensive simulation. The results of these studies follow.

The typical wind speed is given as V_W = 10 kt from 240° (fig.13). On the basis of this relatively low wind velocity, the wind model shown in figure 4 was developed. The different wind shear profiles result from the variable surface "roughness" of the area - (a neutral stability of the atmosphere was assumed). At higher wind speeds, additional eddies will take place but, with the present simulation model, these values are not sufficiently accurate [10]. The maximum downdraft velocity at a median west wind of V_W = 10 kt is 0.25 m/s.

The effect of orographic induced wind shear and downdraft is especially serious during take-off and missed approaches at airports, where the take-off and landing weight is limited due to runway length and obstacle clearance. This influence shall be demonstrated on a twin engined aircraft with an engine failure at the critical take-off speed V_1 or at the decision height.

For comparison, fig.14 shows the take-off and climb paths of a typical twin-engined medium-haul jet aircraft (A) and a typical twin-engined short-haul jet aircraft (B) in a case where second segment climb is hindered by an obstacle in the flight path. In this case, the flight path determines the maximum allowable take-off weight or, obstacle-limited take-off weight. The limited allowable take-off weight under these conditions is shown as a percentage of the aircraft's structural weight. Despite the different take-off weights and engine thrust the flight paths are essentially identical. From the standpoint of flight mechanics, the procedure can be described as "standard".

In the case of a limited take-off weight because of obstacles, the flight path is directly affected by the terrain and obstacles. In the case of a go around or a missed approach, however, the FAA does not stipulate a corresponding consideration of the terrain under the probable flight path. The rules here, assuming an engine failure, require only that a minimum climb gradient of 2,1% be maintained. Nevertheless, when the decision for a go-around is reached, obstacles within the missed approach area must be considered. In the event of an approach followed by a go around,where the engine failure occured prior to the beginning of the landing approach and the landing flaps have been set accordingly, it is assumed that the flight path always will be over areas known to be obstacle-free. If engine failure occurs at decision height while the aircraft is in a normal, both-engine landing configuration, it can (under certain circumstances) fly through the obstacle-free zone within the missed-approach area. The reason for the latter is the need of the aircrew to get the flaps from the "normal" landing position to the configuration for an engine-out balked landing. This takes time and has the effect of "stretching" the badly needed horizontal flight path for acceleration to a safe control speed or risking a loss of altitude. Despite identical points for the initiation of the missed-approach procedure, the performance characteristics of different aircraft result in different flight paths as shown in fig.15. Low-performance aircraft pose more of a problem.

This problem has been discussed for years in the International Civil Organization's (ICAO) Airworthiness Committee without any effect - so far - on airworthiness. Nor has the revision of the existing Procedures for Air Navigation Services - Aircraft Operations (PANS-OPS) brought any progress in this context. The new PANS-OPS contain, among other things, new and modified methods for the determination of obstacle-free criteria. The directives are based upon results achieved with the help of scientific methods including a Collision Risk Model (CRM) which can estimate the risk of a collision with obstacles

under given marginal conditions. The computer model will be put at the disposal of the ICAO member States. The method, however, does not include such "extraordinary" factors as wind shear or turbulence.

A special hazard to flight safety during both take-off and go around can be created when climb capability is degraded by unfavourable wind conditions. Wind has a direct influence on the flight path and, as a result, on the obstacle clearance height. Wind also is a key factor in computing the maximum allowable take-off weight.

Wind speed and direction depend upon both position and time. In other words, the wind factor is a continuous variable affecting an aircraft's flight path. The resulting aircraft dynamics can be computed only with the aid of a costly simulator. To simplify the take-off procedure, the wind is considered to be constant. This relative wind velocity is measured at the surface and a median speed is an average over the last 10 minutes. As far as wind speed for take-off is concerned, the FAA requires that headwinds be calculated at 50% of the nominal value, and performance-decreasing tailwinds considered at 150%.

In "worst-case" situations, the variation between the actual wind conditions and the reported median can be as high as 80%. Wind shear and updrafts or downdrafts are not included in take-off computations despite their effect on the climb-out flight path. The reasons for this is the current lack of measuring capabilities and the long-standing assumption that the vertical wind components near the surface are so small they may be ignored.

Fig.16 shows gross and net flight paths for a wind simulation as indicated in fig.4. (The corresponding flight paths under no-wind conditions are shown in fig.14) The strong influence on flight path displacement even with a relatively low headwind of V_W = 10 kt is quite pronounced. Both aircraft A and aircraft B are well below the 10,7 m minimum clearance height even though, according to the regulations, only 50% of the headwind component is used to compute take-off weight. Without the influence of wind shear and downdrafts, the actual overflight altitude would be much higher than the minimum allowable overflight altitude due to the 50% "risk" factor. But the result is that the 50% "safety" factor, in the given wind model, is completely used up. In the often-underestimated "lee influence", the low flight path under engine-out conditions can be wiped out by a downdraft velocity of 0.12 m/s despite the improved climb performance provided by a headwind of 10 kt. Additional air recirculation or variations in the measured wind speeds would reduce obstacle-clearance even more.

Despite the relatively comparable take-off performance of the two hypothetical aircraft (fig.14), the two aircraft have very different flight paths under the wind conditions studied (fig.16). The reason is in the dynamic reaction of the aircraft to the variable wind conditions. An important parameter here is the thrust radius - the geometric distance between the engines and the aircraft's center of gravity (CG) - and the relationship between the thrust available and airspeed. A study of other twin-engine aircraft indicates that the farther below the CG the engines are mounted, the stronger and, in general, less favorable are the aircraft's dynamic reactions in its flight path. This "problem", however, should not be overestimated in a discussion of obstacle clearances.

For low-performance aircraft, a one-engine go around can be far more critical than a take-off (fig.17). Under the wind parameters shown in fig.4, aircraft B would hit the obstacle. Under still-air conditions, the clearance height would be marginal, but would meet regulations.

4. OPERATIONAL CONSIDERATIONS TO PREVENT WIND SHEAR ACCIDENTS

As the response of a piloted aircraft differs very much during approach and landing and on the other hand during take-off and go around, the procedures for accident prevention are quite different.

TAKE-OFF

A trivial but powerful advise would be: don't start under thunderstorm conditions. The probability to survive is small in strong thunderstorms during take-off.

The inclusion of wind shear considerations excluding thunderstorms in take-off and go around regulations makes sense only when related to specific obstacle conditions and when certain weather situations can be predicted with some probability. It also would be necessary to compute, with a simulator, the actual situation. The simulation model would have to include actual wind values as measured at the airport in question. This technique could be supported by information to pilots that certain wind situations could occur at certain airports and the pilots should be alert for such possibilities. At new airports, another possibility would be the consideration of the statistical distribution of the main wind directions as well as of the surrounding terrain's effect on the wind. Existing airports could be modified to minimize any restrictions on flight operations.

The reduction of flight weight to provide a great margin of safety for take-off and go around - especially in the case of unforeseen wind shear - is an economic factor which no one regards with pleasure. For take-off, it has been shown that the 50% headwind factor provided by the FARs can be completely consumed under the conditions of the hypothetical wind model presented here. Other areas such as inexact wind velocity measurements and time-variable winds cannot be covered by the 50% safety factor. It would seem here, that an increased take-off weight due to a headwind component should not be allowed under the conditions described. It has been shown that the FARs for go around procedures impose far lower requirements than for take-off. This is most likely based upon the fact that, when a go around decision is made, the altitude and course already exist and the process is somewhat less critical.

If, however, an engine failure is assumed at go around decision height - similar to the V_1 decision speed at take-off - the flight distance required to reach a safe climb speed is quite long. In this case the FARs should require that gross flight paths for go arounds under the marginal conditions noted above should be computed and, as for the net flight path, provide a minimum obstacle clearance of 10.7 m in the "Balked Landing" sector. It is admitted that, according to the examples given, the authorized maximum landing weight would be reduced considerably, but a safety minimum would be guaranteed.

LANDING APPROACH

In contrast to take-off and go around accidents, wind shear accidents during approach and landing could generally be avoided, if the pilot or the automatic flight control system reacts in a correct manner. In all approach wind shear accidents, analyzed by the author, neither limited flight performance nor slow engine response time, nor exclusively non-adequate control of elevator and thrust were the basic reason for the crash.

As mentioned in chapter 3, the total energy of an uncontrolled aircraft in a wind shear situation is nearly constant, the airspeed deviations are negligibly small and the aircraft will be accelerated without significant time delay with the time varying wind. The main deviation occurs on the flight path. In case of an increasing tailwind, the aircraft would be accelerated inertially and this will increase the kinetic energy. If the total energy is constant the potential energy and therefore the height has to decrease. From this unexpected aircraft response results:

- pilot and automatic flight control system can neither indicate a wind shear situation from airspeed deviation nor from criteria based on total energy deviation;
- a powerful wind shear indication is the deviation from the glide path.

It can be assumed that flight safety will be guaranteed in a wind shear situation if airspeed and flight path deviations are small. In this case additional thrust will be required to accelerate the aircraft without flight path deviation.

The required thrust or specific excess power $\dot{H}_E$ to avoid airspeed and flight path deviations is shown in fig.18 curve R for the wind shear gradient of fig.5.

Today's flight control systems based on the classical concept of separating autopilot and autothrottle can already reduce the hazards of wind shear to a considerable amount. In fig.18, curve A, the throttle activities of such an automatic control system of a European wide-body aircraft is given in the above mentioned linear tailwind shear.

A comparison with the required values (curve R) reveals that this modern flight guidance and control system executes principally correct compensations.

The excursions in the airspeed and flight path signals originate from the special flight control structure. A high throttle activity as a response to higher frequency gusts is avoided here by means of a complementary filtering technique. This brings the disadvantage of a delayed counteraction against the low frequency wind shear disturbances only after relatively great offsets of airspeed and flight path have established.

The employment of stronger cross-coupled flight control systems, no longer separated into autopilot and autothrottle and operating on the basis of an energy management [14]leads to a further considerable reduction of the total energy excursions while using conventional sensoring.

By means of an additional direct open-loop compensation the offsets in airspeed and flight path caused by wind shear can be eliminated almost entirely [11]. This method is based on the on-board measurement of the wind vector components and their time derivatives and a corresponding thrust command signal. In this way an ideal thrust signal in wind shear is generated (see fig.18, curve R) without changing the original stability of the controlled aircraft. Two main disadvantages have to be noted, however: first, complete airdata and inertial data must be available on board the aircraft in order to determine the wind components. Secondly, the above mentioned complementary filtering of the wind signal can no longer be maintained because it would counteract the open-loop activation of the thrust due to its structural composition. This method cannot be employed successfully until the problem of separating the wind shear and gust signals has been solved completely.

A management of the aircraft's energy and energy rate (based on airspeed) leads despite its simple structure to very small deviations of the airspeed and height. The concept for such a flight control system can be derived from the following principle: The required thrust is proportional to the deviation in specific excess power $\tilde{\dot{H}}_E$, and the time integral of $\tilde{\dot{H}}_E$, the energy height error $\Delta\tilde{H}_E$, is an indication of the total energy state of the aircraft [1]. The specific energy rate deviation can be calculated simply by a linearized approximation:

$$\Delta\tilde{\dot{H}}_E = - \frac{V\,\dot{V}}{g} + V\,\Delta\gamma + \gamma\,\Delta V$$

with: V airspeed

γ flight path angle

- : deviation
- • time derivative.

In order to determine this equation, no expensive measurement is necessary. It is sufficient to combine existing sensor signals. Curve B in fig.18 shows the answer of the specific energy rate and the corresponding state of flight variables when the principle of the total energy rate management is applied. The unimportant deviations of the airspeed and height are substantially referred to the influences of engine time delays. Nevertheless, filtering of the V- and $\dot{V}$-signals is still necessary in order to prevent the thrust from following each gust.

Great difficulties for pilots occur in downdraft or downburst situations. To maintain flight safety the pilot should keep angle of attack, airspeed, and flight path angle constant. In case of a downdraft the pilot has to pull the control column in order to increase the aircraft's pitch angle [14]. The procedure to pitch up the aircraft in a situation, where the airspeed decreases (although only slightly), is adverse to the pilots feeling and training. As downdraft counteraction isn't generally implied in most flight director control laws, the pilot tries to keep the pitch angle constant with the result of an undesired flight path deviation. Therefore pilots should be trained to react adequately in downdraft situations.

WIND SHEAR WARNING DISPLAY

Concerning the prevention of wind shear accidents the major problem which has to be solved is to inform the pilot in a proper manner about his situation. In principle all sensor signals the pilot needs for a proper information are available in the conventional cockpit instrumentation. Even in well equipped wide body aircraft the pilot is not yet able to correlate all of these informations in order to realize a correct warning.

Wind shear can be described by characteristic values, of the shear gradients, the thickness of the shear layer or the overall shape of the wind changes. On board the aircraft only the momentary wind gradients can be determined. But note that in contrary to the opinion among experts [13] the gradients alone are no exclusive measure for the hazard. No forecast of the expected total event of the wind shear can be given based only on the knowledge of the momentary wind shear gradients. And there is yet no evidence that an aircraft is more endangered by a high wind gradient, lasting only a short time, than by a small but persistent gradient which is possibly not recognized by the pilot. A better means for evaluating the threat of a wind shear to the aircraft appears to be an energy height error. As far as we could investigate, energy height errors of the magnitude of 15 to 20 meters (resp. kinetic energy errors of around 2.5 m/s) may be tolerated at higher altitudes during take-off and landing. However, the allowable errors have to be obviously narrowed with decreasing height of the aircraft above the ground. To avoid uncomfortable and unnecessary miswarnings, the pilot should not be warned until these limits are violated.

It appears difficult to supply the pilot with another information in view of the great burden of control task he has in a landing approach. The question arises whether to install additional instruments or to modify already existing displays. This is more or less a question of philosophy that is certainly going to answer itself when new or modified instruments fulfil the one and only requirement: They must display the proper quantity that will only warn the pilot when it is necessary, and that will give him appropriate guidance when he needs it.

The concept of displaying airspeed based on energy and energy rate [1] (chapter 4) has been tested in a moving cockpit simulator by a joint team of Bodenseewerk Geratetechnik, Deutsche Forschungs- und Versuchsanstalt für Luft- und Raumfahrt (DFVLR) and Technical

University Braunschweig. This research was sponsored by the German Ministry of Transportation (BMV) [6]. Fourteen airline pilots flew different approaches, where the New York thunderstorm profile [7] (fig.10,11) caused the heaviest problems. In the absence of additional aircraft systems, e.g. autopilot wind shear warning, none of the well motivated pilots were able to land the simulated wide body aircraft without a crash, even though the response of experienced pilots was different from less experienced pilots. (curve 1, fig.10,11)

With the display of energy and energy rate, most of the pilots recovered in a hard but safe landing (fig.10,11, curve 2). During these simulator studies the questions arose, if it is worthwhile to display energy and energy rate in different instruments or to combine both signals in one display [12]. This question shall be answered in an additional simulator study, where man-machine problems will be optimized. The main results of the simulator studies were:

- pilots (both well or less experienced) are not able to make a safe landing under severe wind shear conditions without additional support of an automatic flight control system or an adequate wind shear warning display;
- if there is enough training available, pilots can adapt themselves to specific wind shear profiles. It is therefore necessary to expose the pilot to different wind shear situations. A general ground based wind shear warning is worthwhile but not sufficient;
- an adequate wind shear warning display can support the pilot in the most severe wind shear situations.

5. CONCLUSIONS

Wind shear during take-off, go around, and missed approach is a pure flight performance problem. Pilots should avoid to take off into thunderstorms. Moderate wind shear induced by orographic lee effects can be overcome by increasing the thrust to weight ratio, especially in engine failure conditions. In unexpected dangerous situations the pilot is advised to reduce the airspeed safety margin in order to increase the obstacle clearance. Wind shear accidents during landing and approach could generally be avoided if the pilot keeps the automatic flight control systems in operation and if he is informed by an adequate wind shear warning display. Wind shear is particularly dangerous if it occurs in a height of approximately 80-120 m.

A ground based wind shear warning is worthwhile but not sufficient. The adequate information of the aircraft response in wind shear can only be measured on board. A major parameter is airspeed; high airspeed leads to greater flight path deviation.

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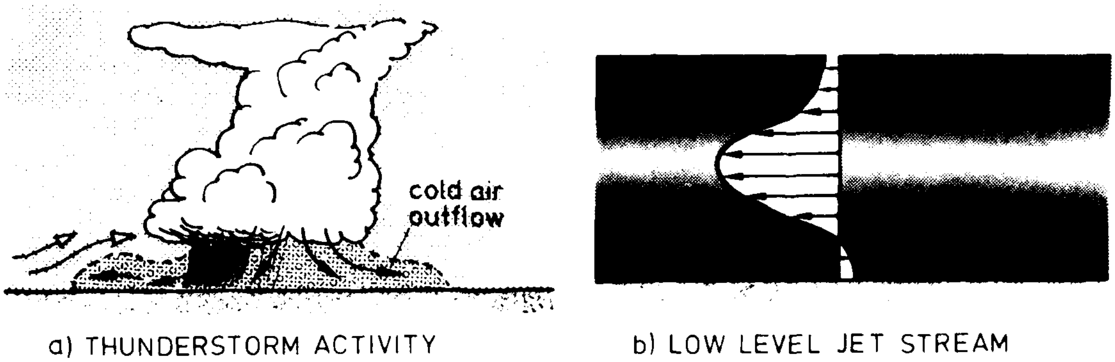

Fig. 1: Meteorological Scenario significant for wind shear conditions. [1]

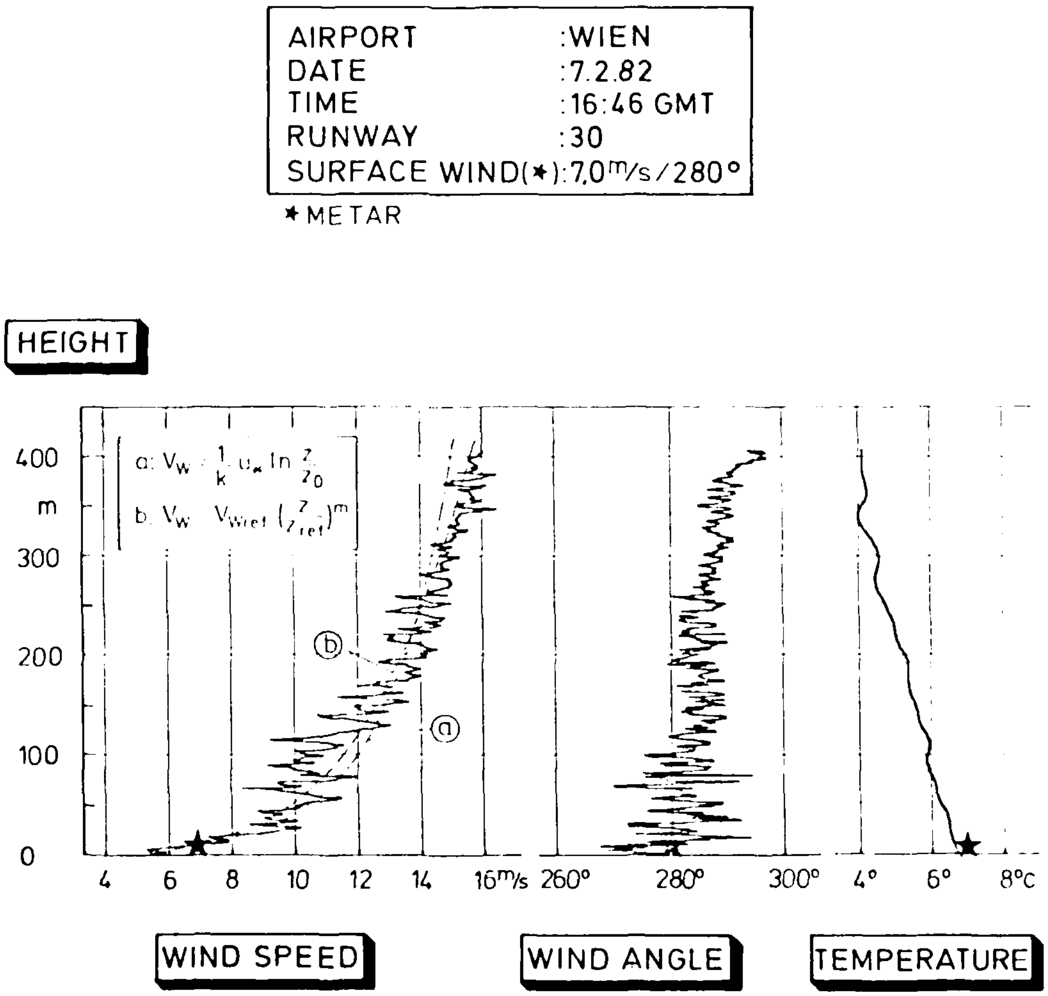

Fig. 2: Example of a typical boundary layer wind profile.

AIRPORT	:FRANKFURT
DATE	:12.10.80
TIME	:14:26 GMT
RUNWAY	:07
SURFACE WIND(*)	:5,5 m/s/010°

*METAR

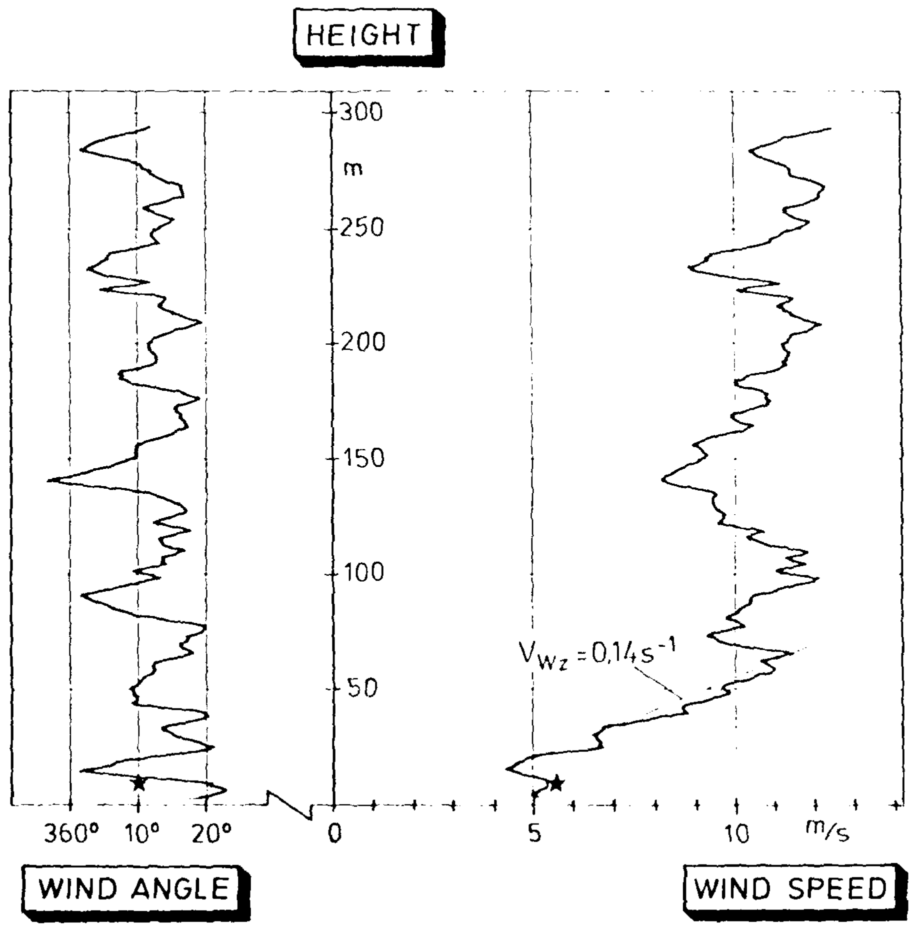

Fig. 3: Sample wind shear profile

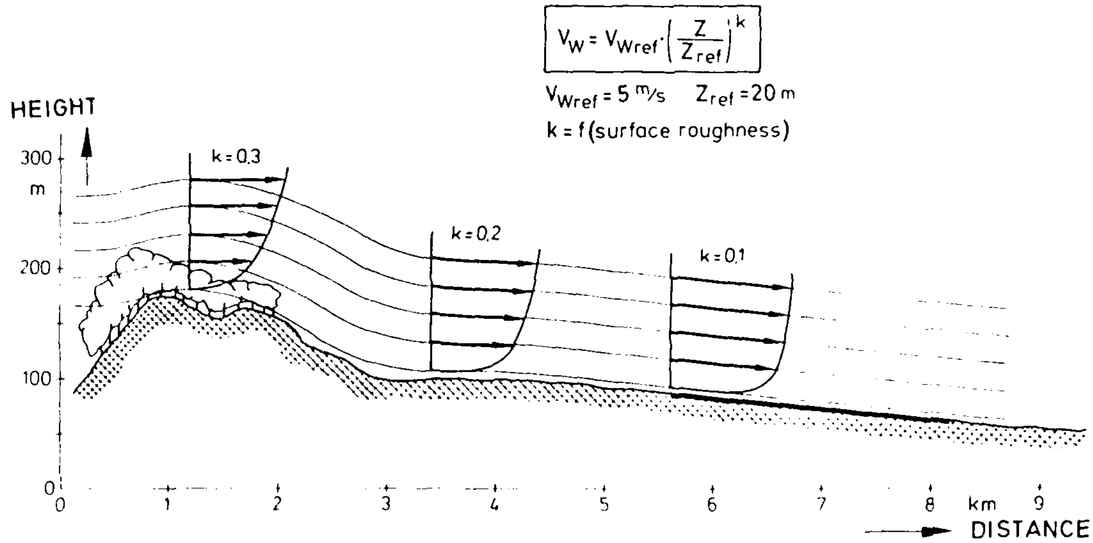

Fig. 4: Model of windstream lines and profiles over a mountain ridge [3]

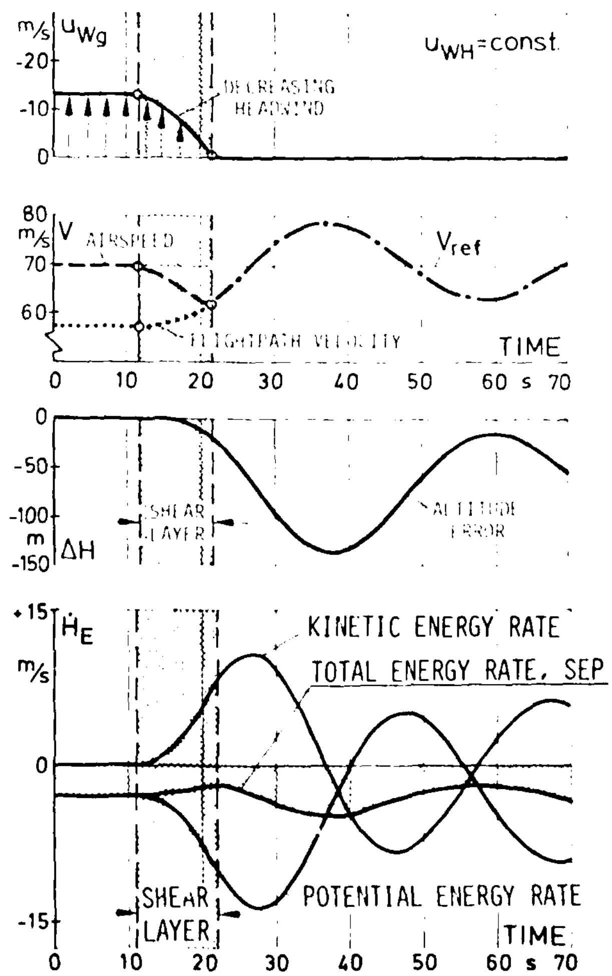

Fig. 5: Effect of tailwind shear on velocities, altitude and energy deviation.

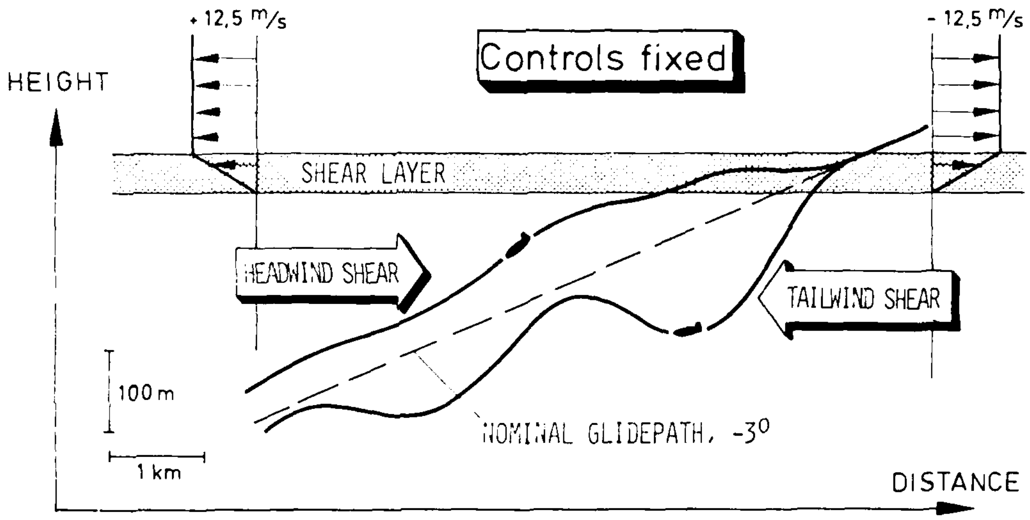

Fig. 6: Landing approaches in headwind resp. tailwind shear, controls fixed [1]

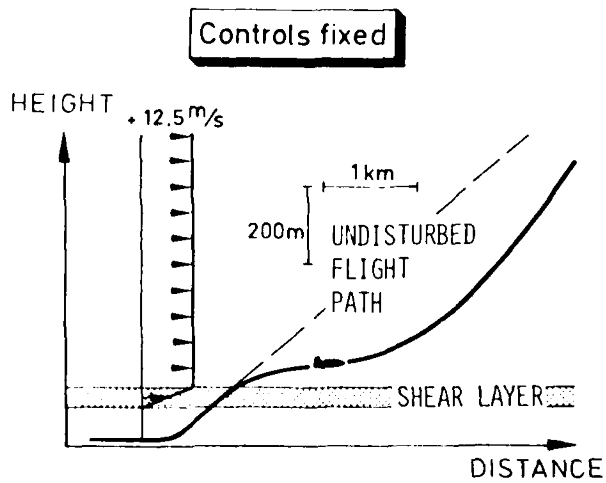

Fig. 7: Take-off flight path in tailwind shear, controls fixed [1]

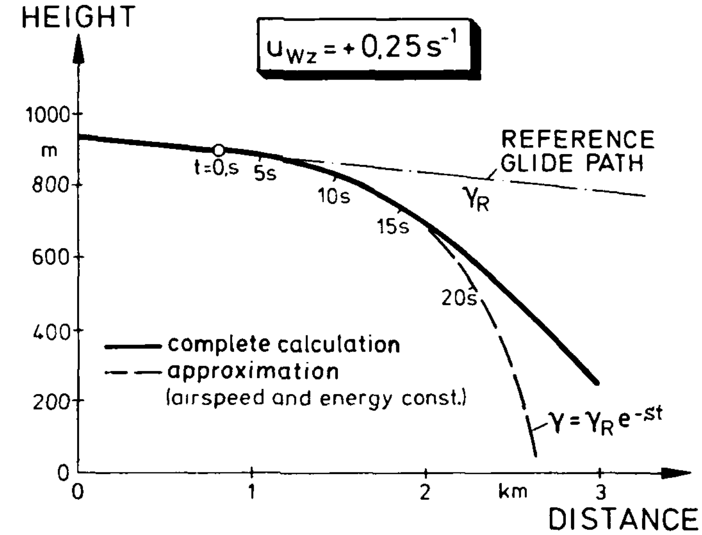

Fig. 8: Flight path of an aircraft with fixed controls in a constant vertical wind shear gradient u_{Wz} [5] (s = Eigenvalue of phugoid mode, see fig. 9)

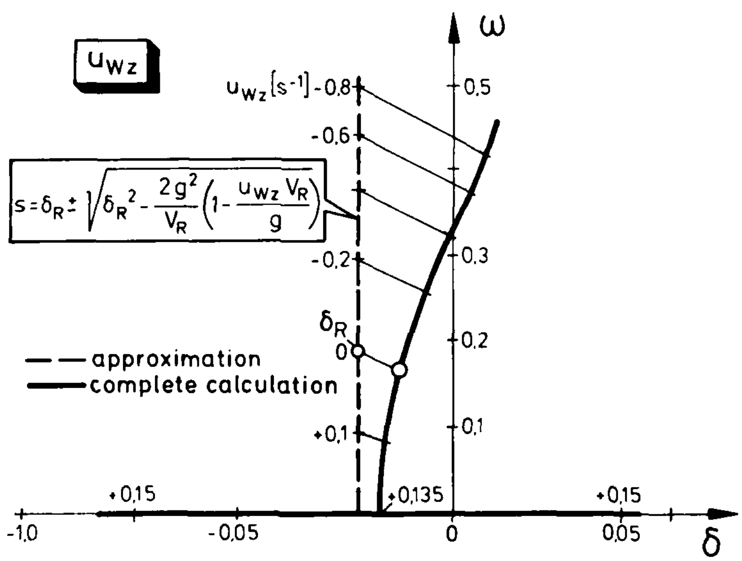

Fig. 9: Eigenvalues of the phugoid mode as a function of a constant vertical wind shear gradient u_{Wz} [5] (Index R: Referencedata)

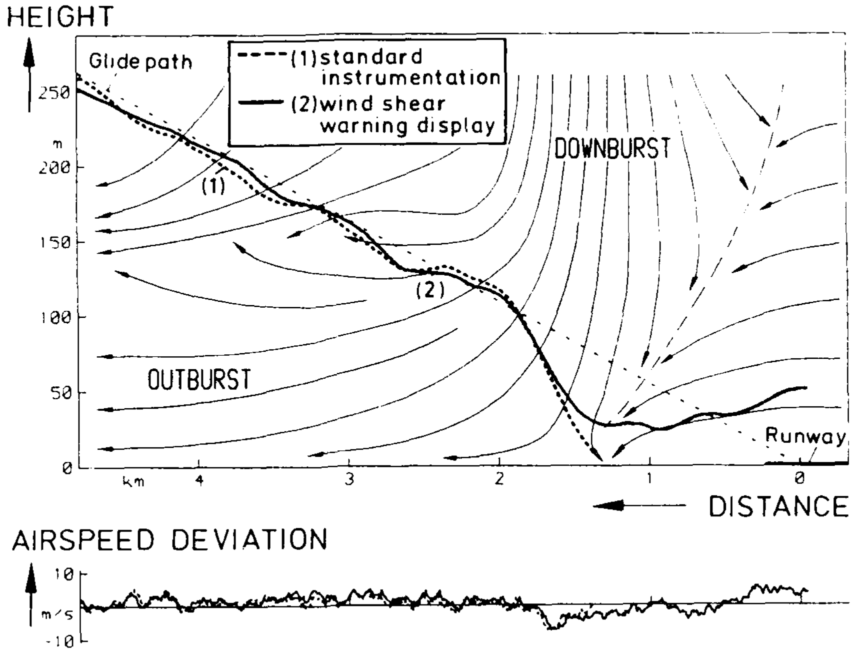

Fig. 10: Flight simulator approach in wind shear conditions, experienced airline pilot

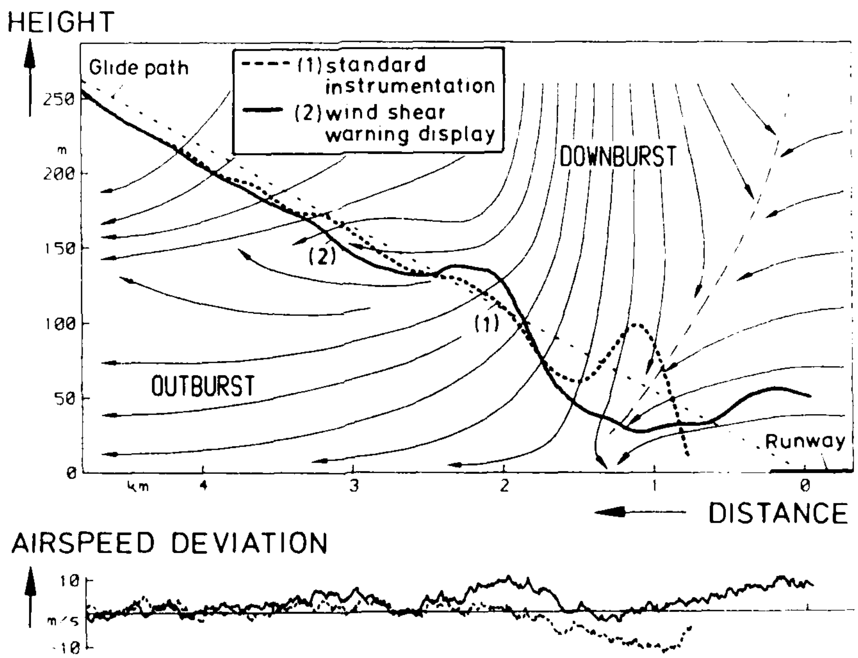

Fig. 11: Flight simulator approach in wind shear conditions, less experienced airline pilot

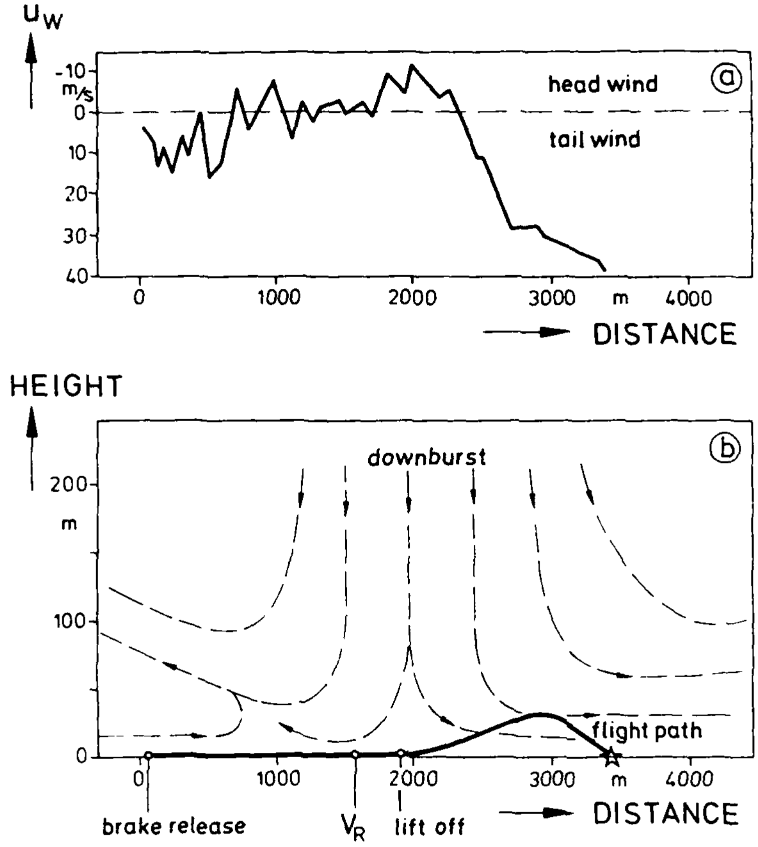

Fig. 12: Analyses of wind shear effect in an aircraft takeoff accident:
a) Horizontal wind component along the flight path; b) Reconstructed traces of windstream lines and aircraft flight path [9]

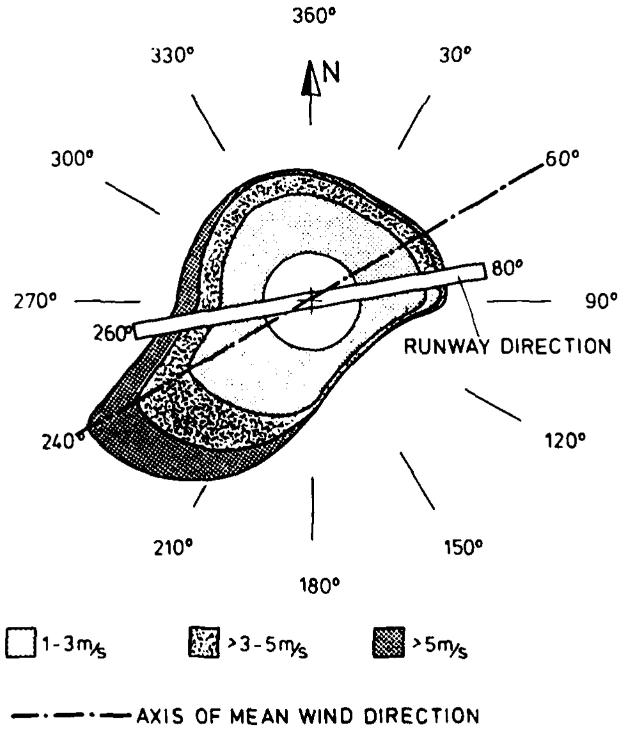

Fig. 13: Annual mean wind distribution of a specific German airport [3]

	take-off weight		
	structure [to]	maximum [to]	limitation [%]
A	137,0	132,0	- 3,7
B	50,3	45,1	-10,3

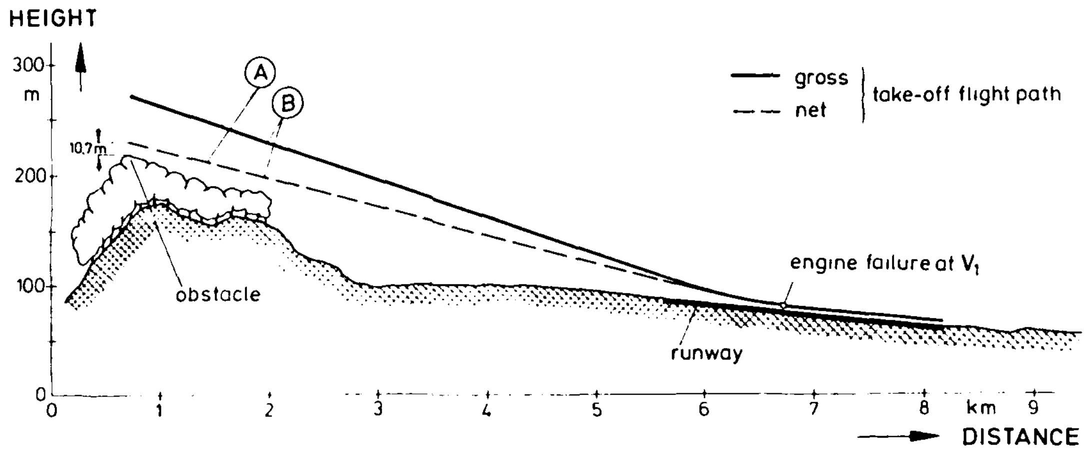

Fig. 14: Takeoff flight paths of two twin-engined jet aircraft under conditions of obstacle-limited takeoff weights and an engine failure at V_1 [3]

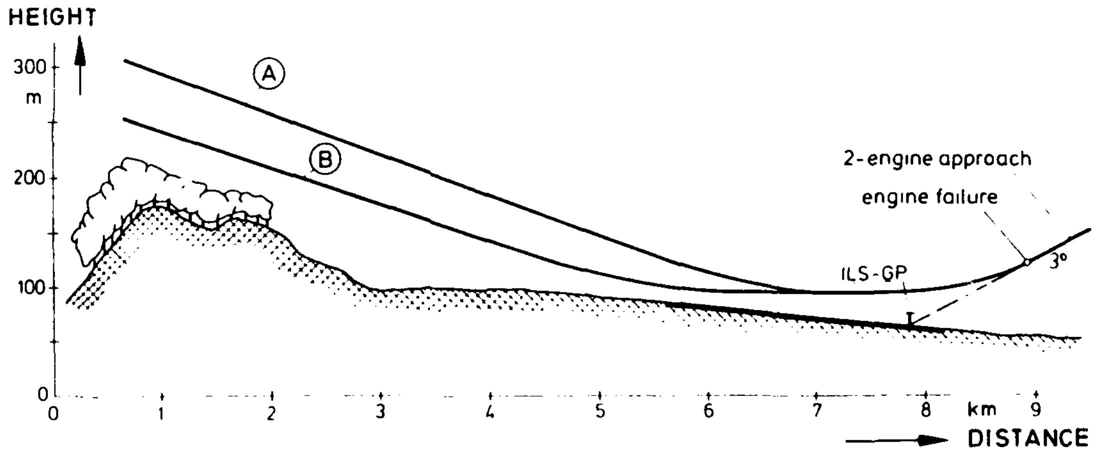

Fig. 15: Go-around flight paths of different aircraft at maximum allowable landing weights [3]

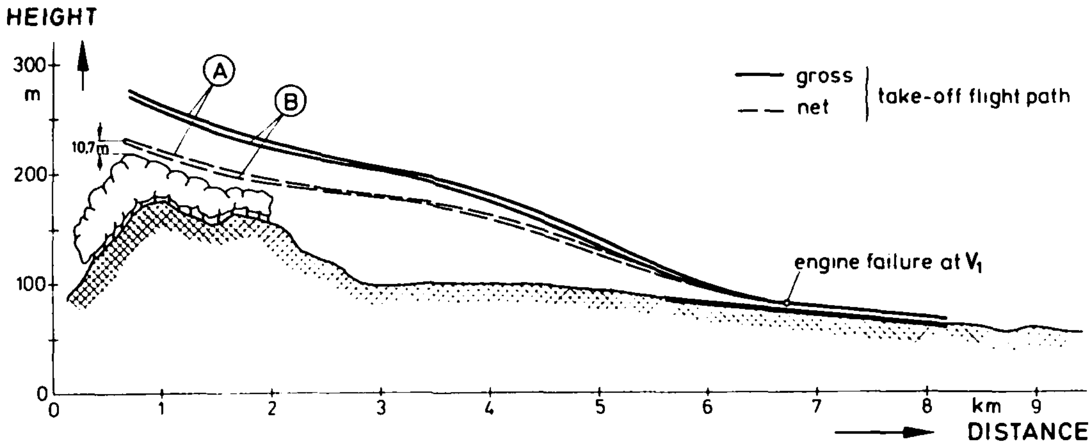

Fig. 16: Takeoff flight paths of two different aircraft in wind shear conditions on the lee side of a mountain ridge [3]

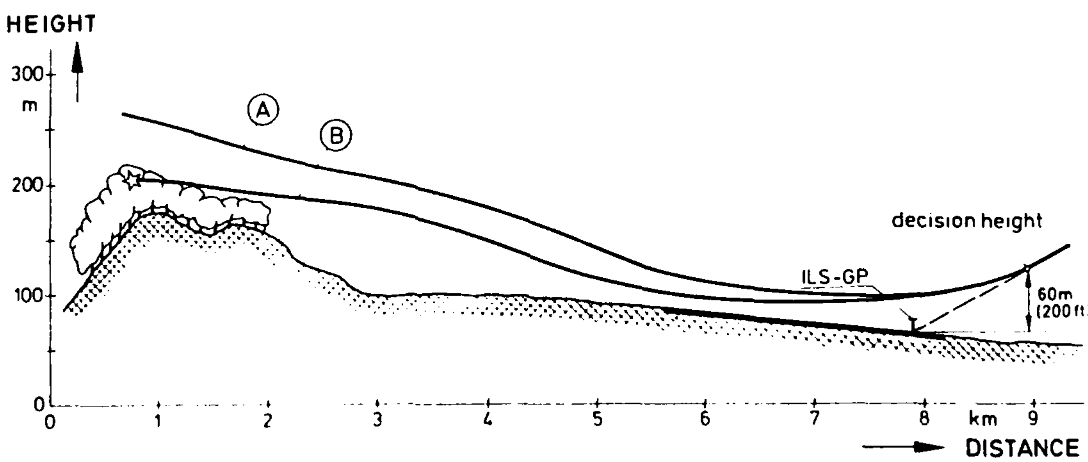

Fig. 17: Go-around flight paths of two different aircraft in wind shear conditions on the lee side of a mountain ridge [3]

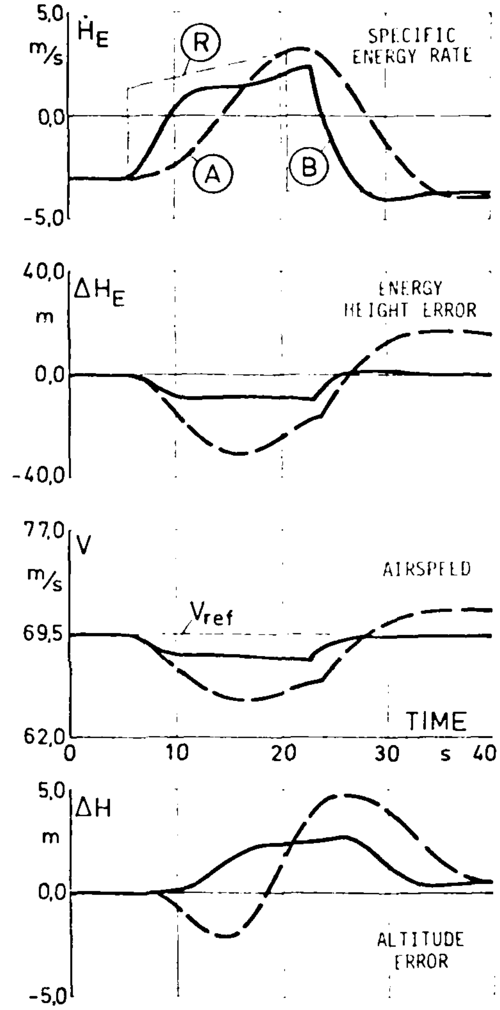

Fig. 18: Effects of flight controls activity in linear tailwind shear.
R: Required specific energy rate
A: Conventional automatic flight controls
B: Specific energy rate management

SOME COMMENTS ON THE HAZARDS ASSOCIATED WITH MANOEUVRING FLIGHT IN SEVERE TURBULENCE AT HIGH SPEED AND LOW ALTITUDE

by

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1. INTRODUCTION

This paper briefly describes the results of some calculations which were made, following several accidents and incidents during service training, in the hope of providing an insight into risks associated with encounters with severe turbulence during high speed manoeuvring flight at low altitude. In that sense, what is discussed here is an attempt to learn lessons from operational experience. But that is not to be taken to imply that the phenomena described are necessarily the cause of any particular accident or incident.

Two dangers will be considered; that a combination of gust and manoeuvre loads may be sufficiently high to cause structural failure, or that this combination may result in a stall or other departure from controlled flight from which recovery is impossible due to the proximity of the ground. In what follows these will first be treated separately, then compared and then an attempt will be made, using turbulence statistics, to indicate the order of probability which might apply. Finally some implications of the results are briefly discussed.

This paper is about high speed low level flight by military combat aircraft. Such aircraft are capable of achieving and sustaining high manoevuring accelerations. This capability is used in terrain following, in manoeuvring to take advantage of terrain screening and for other tactical reasons such as to break the lock of an air defence radar which is tracking the aircraft.

The results presented here relate to an in-service strike fighter at sea level but are not an accurate representation of any particular aircraft. Much of the discussion which follows is over-simplified in order to emphasise the more important aspects and because any attempt at greater accuracy would involve detailed information about specific aircraft which is not available in the open literature. Although the approximations used here are reasonable for present purposes for the class of aircraft concerned, these approximations may not provide a useful approximation to other classes of aircraft and the conclusions drawn should not, without further consideration, be taken to apply to these. All that the present paper attempts to do is to indicate the existence of a possible problem area which appears to have received little attention in the past. It must be for those with a detailed knowledge of a particular aircraft and of the way in which it is used in service, to consider this problem area and determine whether it is likely to be significant in their case.

2. GUST EFFECTS

2.1 Structural Loads. The normal acceleration produced by a gust normal to the flight path of an aircraft varies approximately as the product of the aircraft speed and lift curve slope and inversely as the aircraft mass. This is illustrated, for a typical strike fighter encountering a relatively sharp-edged 15 m/sec gust in level flight, in Fig 1, which shows the total load rather than the load increment due to the gust. Such an aircraft would be cleared to some 8 or 9g at combat mass, with a safety factor of about 1.5. It is apparent from Fig 1 that a very large gust indeed would be required to break such an aircraft if the gust were encountered during straight and level flight.

Since the normal acceleration produced by a gust varies approximately inversely as aircraft mass, it follows that the actual structural load which it produces is almost independant of the aircraft mass. The variation of the combined normal load produced when a 15 m/sec gust normal to the aircraft flight path is encountered during a manoeuvre is illustrated in Fig 2. The load increases strongly with aircraft mass at constant manoeuvre acceleration. However, as illustrated in Fig 3, the total combined gust and manoeuvre acceleration, which is what the pilot feels, decreases at constant manoeuvre g as mass increases. As indicated earlier, at constant aircraft mass the normal acceleration and the load produced by a gust is approximately directly proportional to airspeed, as illustrated in Fig 4.

2.2 Handling Effects. For the typical strike fighter considered here, the change in incidence produced by a relatively sharp edged 15 m/sec gust and the resulting change in lift coefficient are shown in Fig 5. Both incremental incidence and incremental lift coefficient decrease as airspeed increases, both being approximately inversely proportional to airspeed. The lift coefficient required to produce a given manoeuvre acceleration at combat mass and its variation with airspeed is shown in Fig 6. In this case the lift coefficient varies inversely as the square of the airspeed. The lift coefficient produced by a combination of gust and manoeuvre is shown in Fig 7. Aircraft of the kind considered here are often unable to achieve and sustain values of lift coefficient as high as those shown

at the higher values of mass and/or lower values of speed. Undesirable handling characteristics such as stall departure, wing rock, nose slice etc may be encountered at these or lower values of lift coefficient. Since the situation under consideration is one in which the aircraft is flying at high speed close to the ground, encounters with phenomena of this kind seem likely to be disasterous.

Comparing the values of lift coefficient shown in Fig 7 with the values of structural load shown in Fig 2, it appears that encounters with severe gusts during manoeuvring flight are more likely to result in handling difficulties than in direct structural failure. Before attempting to reach a conclusion, however, it is necessary to examine some transient aerodynamic phenomena and this is done in the next section.

The gust induced acceleration and so the bumpiness of the ride felt by the pilot, increases with speed, as shown in Fig 8. However, when flying in turbulence, speed reduction to improve ride quality or perhaps to assist weapon aiming, greatly increases the likelihood of encounters with the handling problems referred to above, unless the amount of manoeuvre acceleration used is reduced appropriately. For the aircraft considered here, the available manoeuvre acceleration at combat mass if an encounter with a 15 m/sec gust is not to result in lift coefficients in excess of 0.8, 1.0 and 1.2 is shown in Fig 9. The rapid reduction of available manoeuvre acceleration as airspeed is reduced is readily apparent, as is the effect of variation of the value of lift coefficient which is selected as that which is not to be exceeded.

2.3 Transient Penetration of Manoeuvre Boundaries. It is well known that values of lift coefficient significantly greater than those of the steady-state still can be achieved when normal acceleration is applied rapidly to an aircraft. Much less is known about what occurs when a wing is stalled by encountering a relatively sharp-edged gust which increases the incidence to a value above that at which the steady stall occurs. Ref 1, although concered with a somewhat different question, gives some idea of what happens. Fig 10 sketches the kind of flow which occurs when an infinite wing of moderate or lower sweep encounters a sharp-edged gust aligned with its leading edge. A vortex with axis parallel to the leading edge is shed from it and moves back over the wing producing a suction peak over the top surface as it goes. The existence of this vortex can be readily demonstrated by placing the hand in a fast moving stream (of water) and changing its incidence quickly.

In addition to generating lift, as it moves over the wing the vortex produces a large and rapidly varying pitching moment. Another such occurs when the gust, followed somewhat later by the vortex, passes over the tail of the aircraft. On a swept wing aircraft a pitching moment will also be generated as the wing becomes immersed in the gust.

The dependance of the behaviour of the flow on how sharp-edged is the gust is not clear, even in this simple case; nor are the 3-dimensional effects when a swept wing aircraft encounters a gust front normal to the aircraft's flight path. The real situation is also likely to be complicated for a variety of other reasons. The gust is unlikely to be aligned at right angles to the aircraft flight path nor is the gust likely to have a component only in the plane normal to it. Thus large forces may be generated which disturb the lateral as well as the vertical motion of the aircraft and, in particular, large rolling moments may be generated on swept wing aircraft. Secondly for aircraft on which the high lift coefficients are achieved by vortex dominated flows, little appears to be known about such transient gust effects. Thirdly, it is often the case that the limiting lift coefficient or incidence is determined by some phenomenon such as pitch-up, wing rock etc rather than by a conventional stall. The behaviour, in the transient conditions under consideration, of the flow fields which are responsible seems likely to be configuration dependant and little seems to be known about it in general terms. Fourthly, the gust may be localised so that the aircraft traverses it sufficiently quickly that there is insufficient time for the aircraft to diverge significantly from its flight path. The question would then arise as to the speed with which the original flow is re-established. For an aircraft flying at a height of some 30 metres, little comfort can be drawn from this fourth point. However, not all of the effects considered above are necessarily unfavourable.

The achievement of lift coefficients significantly greater than can be achieved in steady conditions may affect the consideration of the relative probabilities of occurance of structural failure and handling problems. It should also be noted that the way in which the high values of lift coefficient are generated may change the structural failure mode which would apply.

The motion of an aircraft following a rapid gust induced penetration of a manoeuvre boundary will be influenced by the behaviour of its flight control system. This is likely to be particularly important if a stall prevention/departure suppression/manoeuvre limiting or other carefree manoeuvring system is fitted. How such a system would behave would depend on the details of its design and method of operation, but it would not necessarily behave in the situation considered here in the same way as it would behave in a pilot induced manoeuvre. If the probability of gust induced penetrations of manoeuvre boundaries is as high as is implied by the present paper it is important that flight control aspects are considered and that they be taken into account in flight control system design. This would be no simple matter. The accuracy with which the aerodynamic forces are known is likely to be much lower than that which the flight control system designer has learnt to expect and it is far from clear that the conventional stability derivative approach can be applied in the present situation.

Perhaps the only safe generalisation which can be made is that in the present situation as in others, aircraft with relatively "soft" manoeuvre boundaries are likely to behave in a more benign fashion than would aircraft for which the manoeuvre boundaries are "hard".

3. OCCURENCES OF EXTREME VALUES OF LIFT COEFFICIENT IN SEVERE TURBULENCE

Ref 2 quotes data from extensive measurements made at low altitudes over mountainous terrain in the western USA. Fig 11 is reproduced from it. Noting that in a hard horizontal turning manoeuvre it is the lateral (with respect to the ground) turbulence component which produces incidence changes on an aircraft and that only gusts in one (rather than in either) direction will produce extreme values of lift coefficient, the lateral gust probabilities of Fig 11 have been used to calculate the average periods for which a given manoeuvre acceleration must be applied before values of lift coefficient of 0.8, 1.0 and 1.2 are achieved at Mach numbers of 0.7 and 0.9 at combat mass and at a Mach number of 0.7 at 0.64 times combat mass. The results are shown in Fig 12.

In severe turbulence with characteristics as indicated in Fig 11, when flying at combat mass at a Mach number of 0.7 lift coefficients exceeding 1.2 are likely to occur on average about once per minute of manoeuvring flight at 7g, with lower values of lift coefficient occuring more frequently, for example lift coefficients of 1.0 occuring on average about once every 3 seconds. To retain the same once per minute probability of occurence, at lift coefficients of 1.0 and 0.8 the manoeuvre acceleration would have to be reduced to 5.8g and 4.5g respectively. At combat mass the frequency of occurence is reduced (or the average time between occurences is increased) by a factor of 10 for every reduction of 1g in the manoeuvre acceleration. A much more dramatic reduction in the frequency of occurence, this time by a factor of the order of 1000, occurs for fixed values of lift coefficient and manoeuvre acceleration when the speed is increased from a Mach number of 0.7 to 0.9. For the present aircraft at combat mass, a lift coefficient of 1.2 corresponds to some 12g at a Mach number of 0.9 and some 8g at 0.7. Again, as mass is reduced the average time between occurences of given values of lift coefficient increases markedly if manoeuvre acceleration is kept constant.

For simplicity, the calculations reported here assume that the aircraft was flying at sea level and the turbulence data shown in Fig 11 has been taken as applying to that altitude. However, although that data was measured at a low altitude with respect to the terrain, the measurements were made over the western USA where the terrain itself is some 5000ft or more above sea level. High speed low level operational training is carried out in this area and it should be noted that the effect of increasing altitude to 5000ft at constant TAS would be to reduce the time between encounters with particular values of lift coefficient by a factor of about 5 if the manoeuvre acceleration remains constant.

4. CONCLUDING REMARKS

To convert the above into an appreciation of how often significant values of lift coefficient are likely to be exceeded in operational service requires two further kinds of information. First, data on the frequency of occurence of turbulence of various degrees of severity for the regions in which the flying takes place. Secondly how much time is spent, on average, at particular values of high manoeuvre acceleration at various airspeeds. In addition to the data shown in Fig 11, Refs 2 and 3 indicate various sources of further data on turbulence intensities and much other data exists. This should be adequate for an initial examination although the available data is unlikely to answer all questions. If any information exists on the second question it is only likely to be available to those who operate the aircraft.

If consideration of what has been presented earlier in this paper is thought to indicate the existence of a hazard which is sufficiently significant that action should be taken to reduce it, and this will depend primarily on the high incidence behaviour of the particular aircraft considered and on the way in which it is flown, the question arises as to what this action might be. Awareness of the need to reduce applied manoeuvre acceleration as airspeed is reduced is perhaps the most important point. Also, severe turbulence as discussed here is usually forecastable and levels of manoeuvre acceleration could be reduced when flying in situations when it has been forecast. Severe turbulence at low altitudes over hilly or mountainous terrain is often localised and so the concept that the pilot will be aware of the severity of the conditions from the seat of his pants and so reduce the manouvre accelerations which he applies would at best provide only a partial solution.

Whatever may be regarded as the degree to which the hazards discussed in this paper apply to particular present operations, it appears that they need to be taken into account in the design of flight control systems, particularly when these are intended to provide a "carefree manoeuvring" capability.

SUMMARY: The relative probability during maneouvring flight at high speed and low level in severe turbulence of structural failure or of the inadvertent penetration of maneouvre boundaries is discussed. It is concluded that for a typical current strike fighter the latter is much the more likely. The complexity of some of the phenomena involved is described and the importance of considering such hazards in the design of flight control systems is emphasised.

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Acknowledgement: The author wishes to thank many colleagues in AGARD and in the UK MOD for the benefit of discussions he has had with them on the subject of this paper; however as noted below, responsibility for the views expressed is his own.

Note: This paper gives the views of the author; it does not carry the authority of the UK MOD.

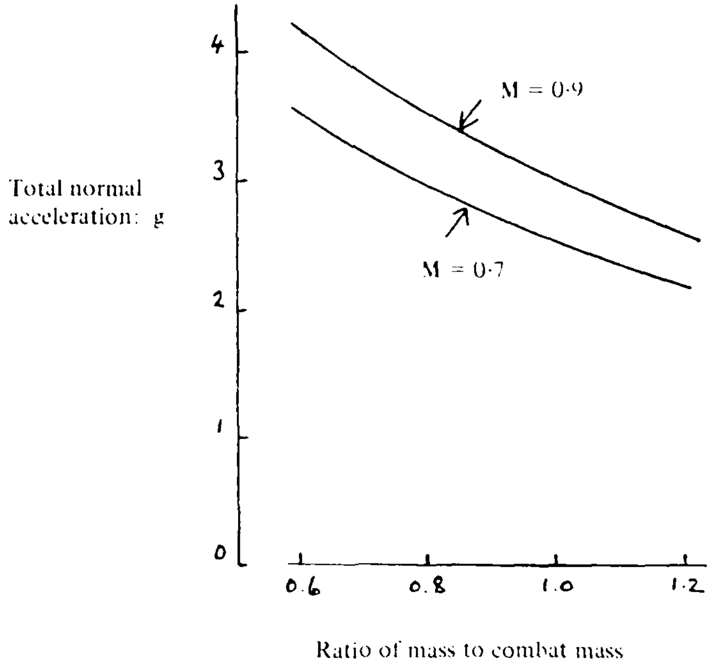


Fig.1 Total normal acceleration when 15 m/sec vertical gust is encountered in level flight

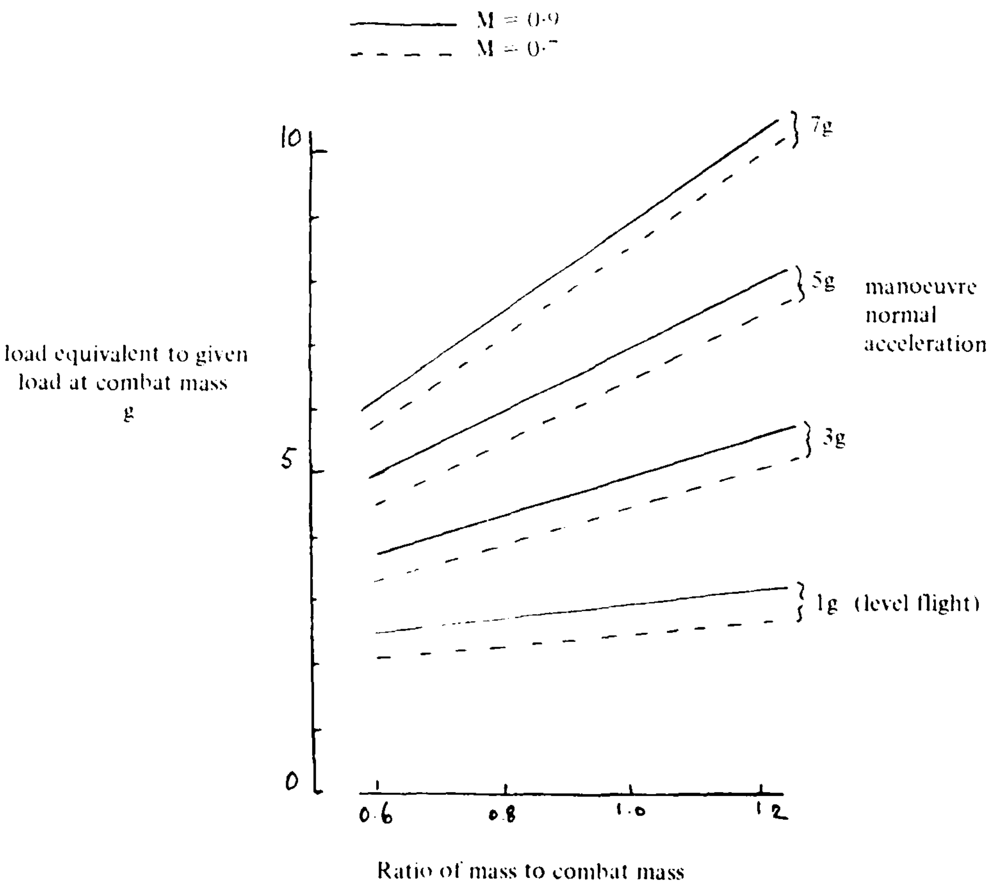


Fig.2 Variation of structural load with combat mass for 15 m/sec gust normal to the flight path encountered in manoeuvring flight

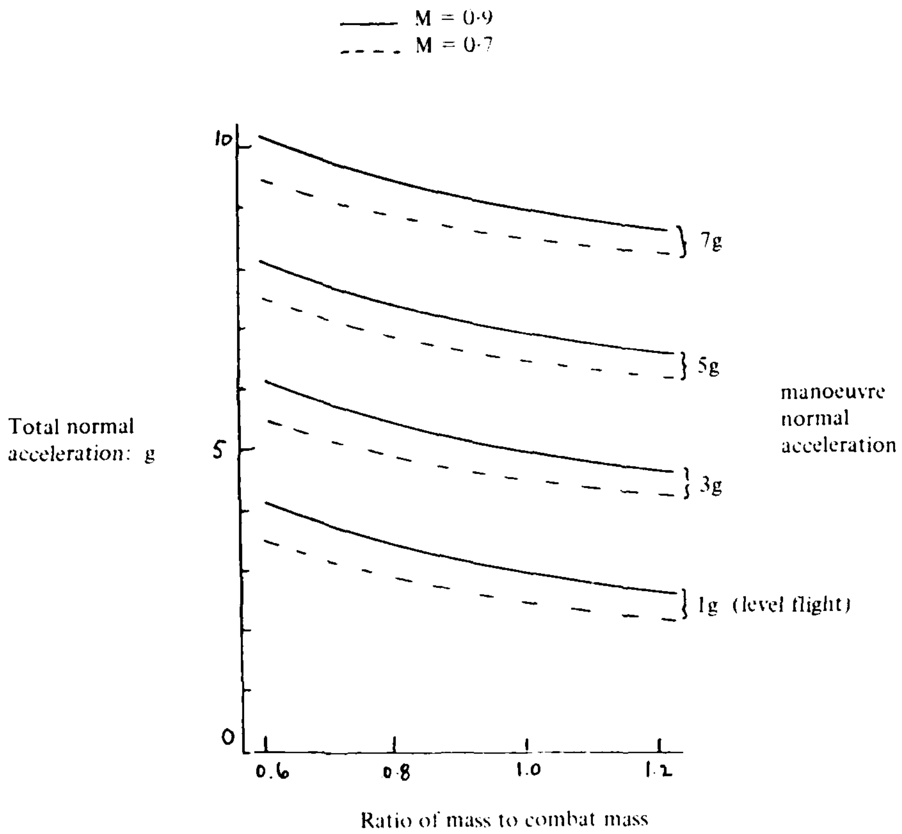


Fig.3 Variation of total normal acceleration with combat mass when 15 m/sec gust normal to the flight path encountered in manoeuvring flight

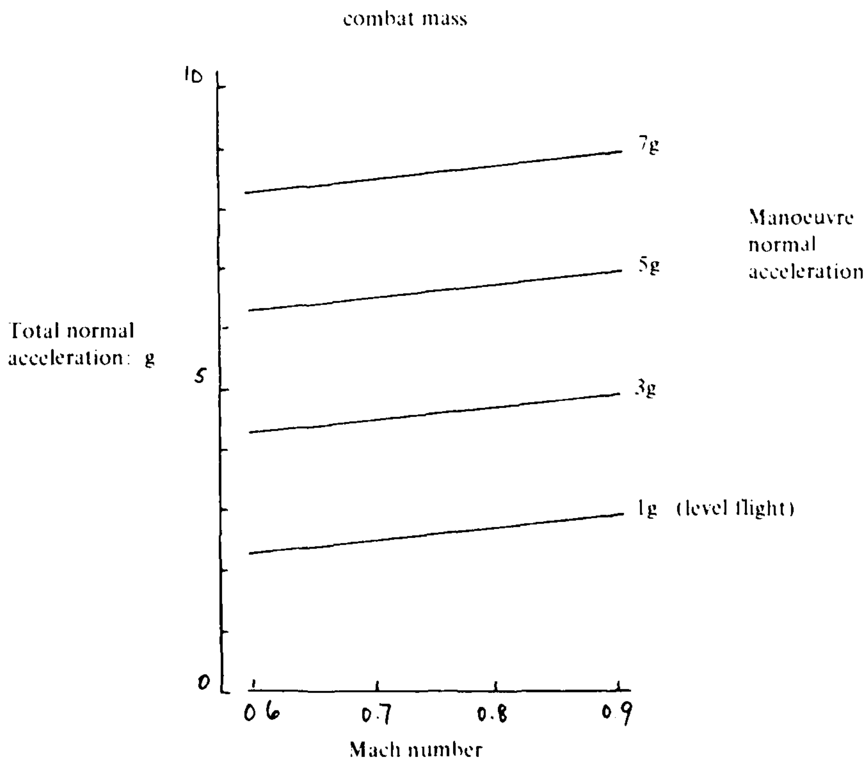


Fig.4 Variation of total normal acceleration with Mach number when 15 m/sec gust encountered in manoeuvring flight at constant mass

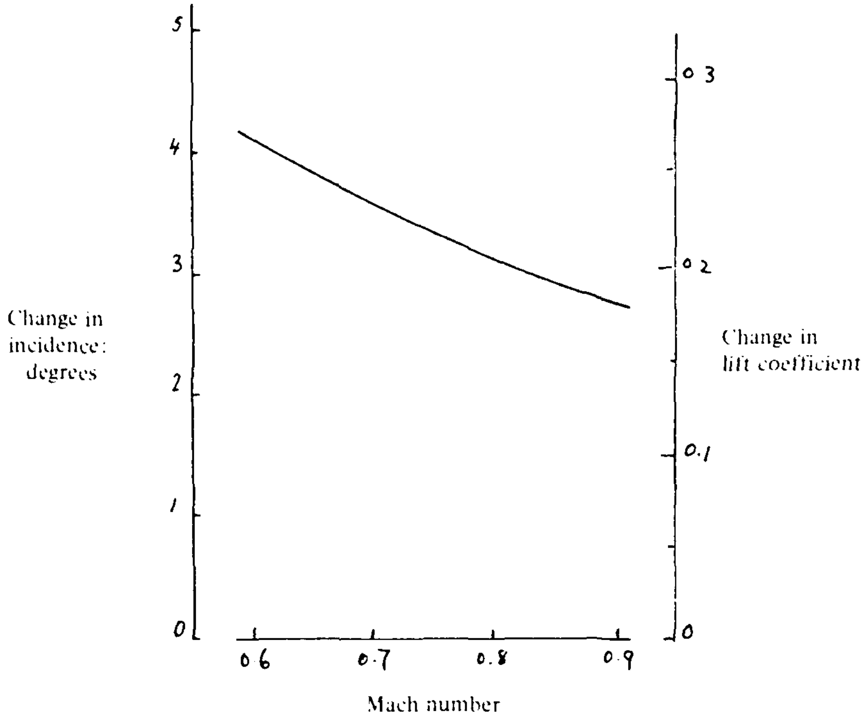


Fig.5 Variation of changes in incidence and lift coefficient with Mach number when 15 m/sec gust encountered at combat mass

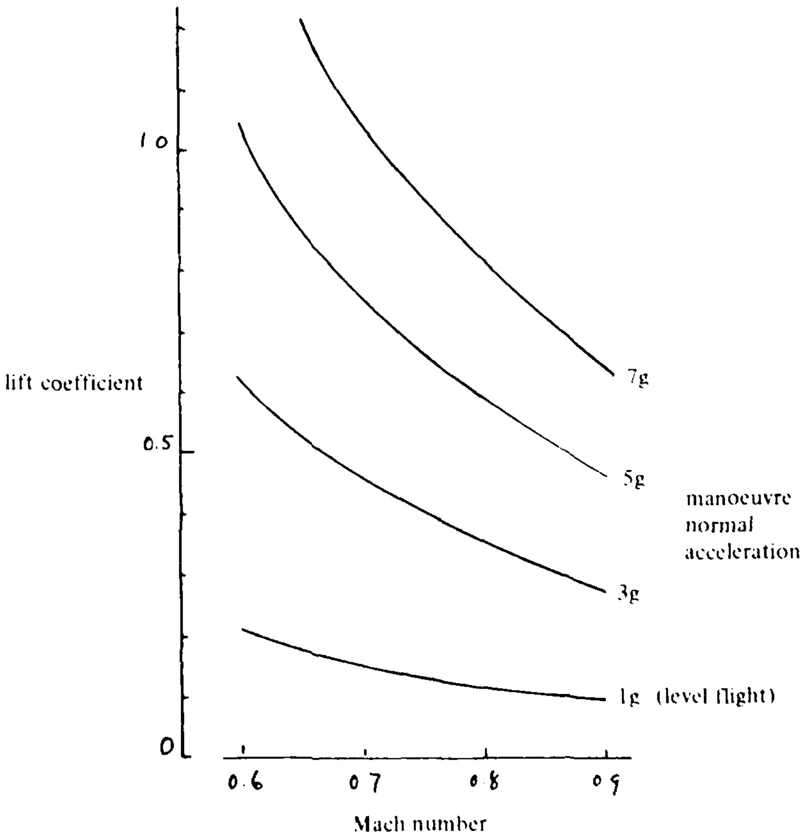


Fig.6 Variation with Mach number of lift coefficient at given manoeuvre acceleration at combat mass

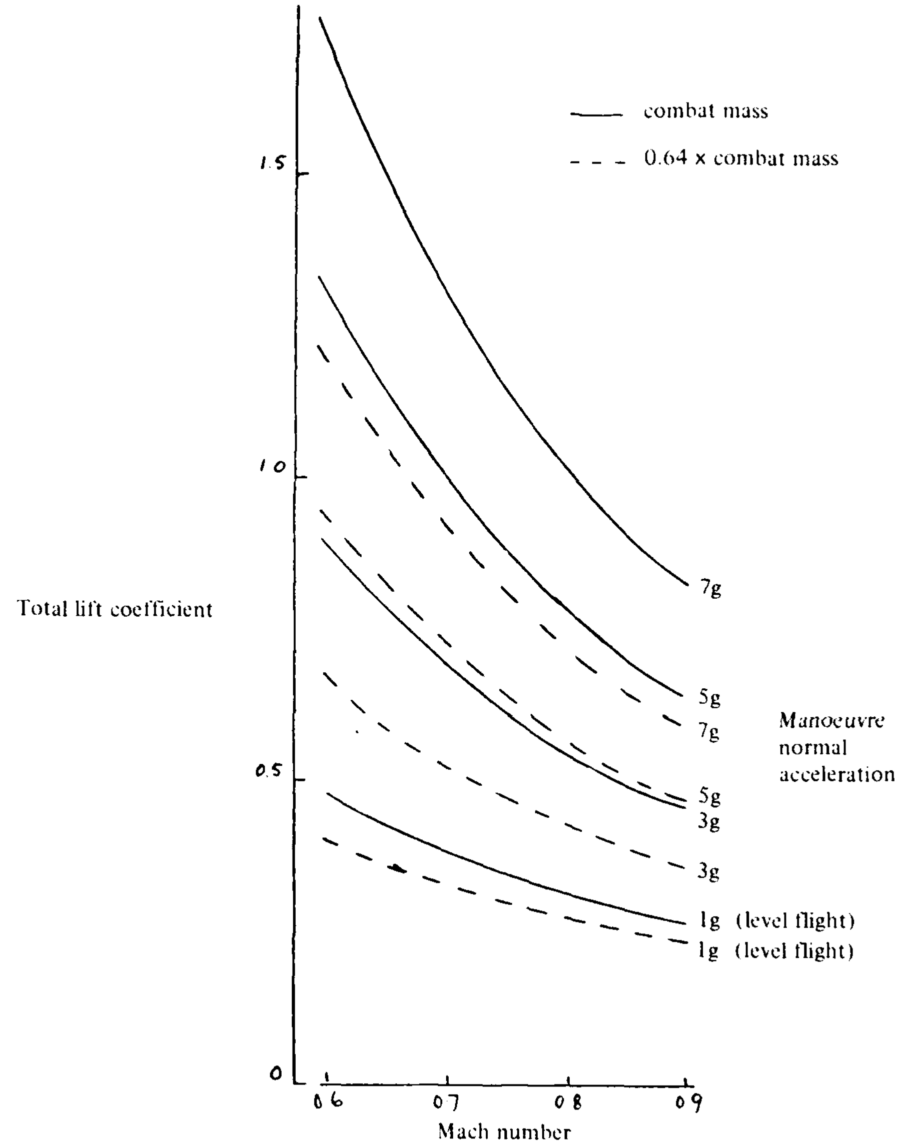


Fig.7 Variation of total lift coefficient with Mach number for 15 m/sec gust normal to the flight path encountered during manoeuvring flight

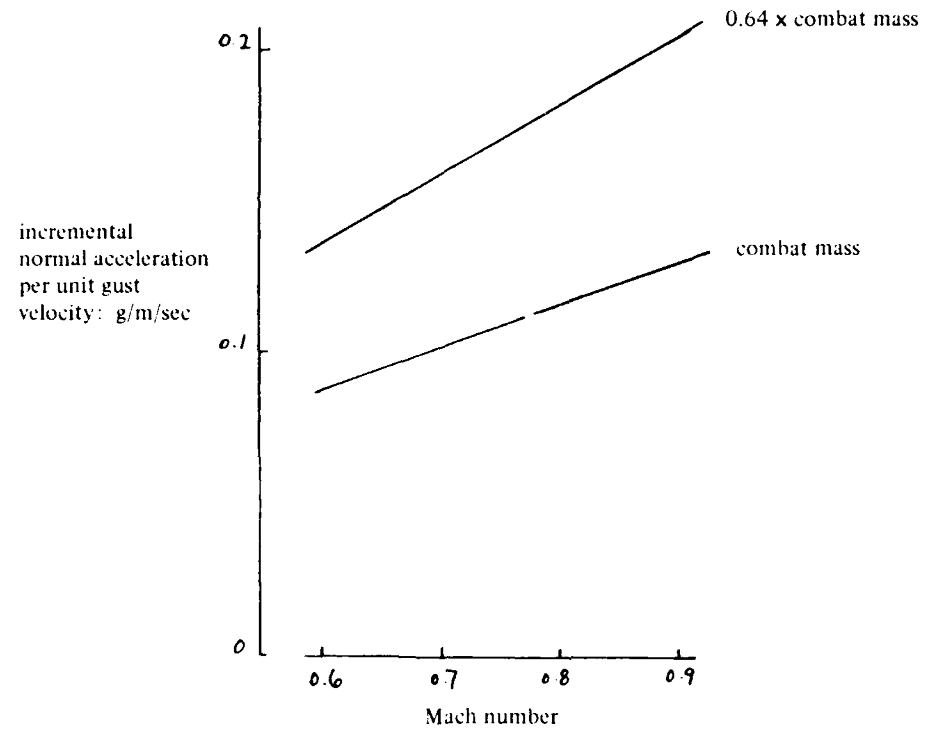


Fig.8 Variation of incremental normal acceleration per unit gust velocity with Mach number

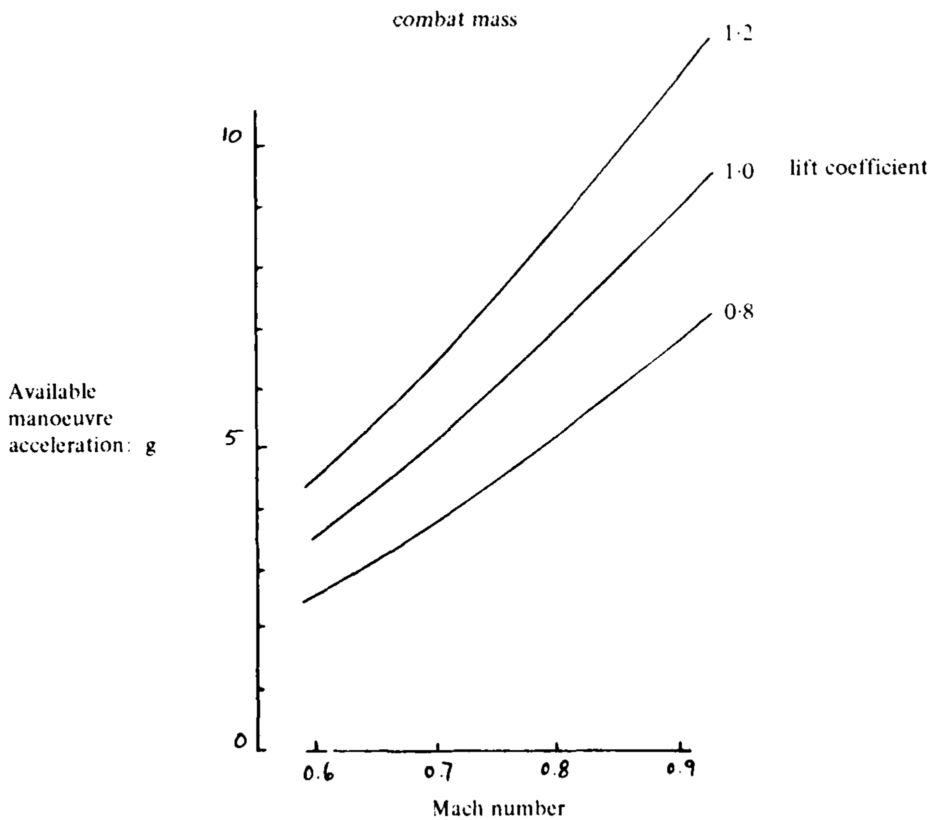
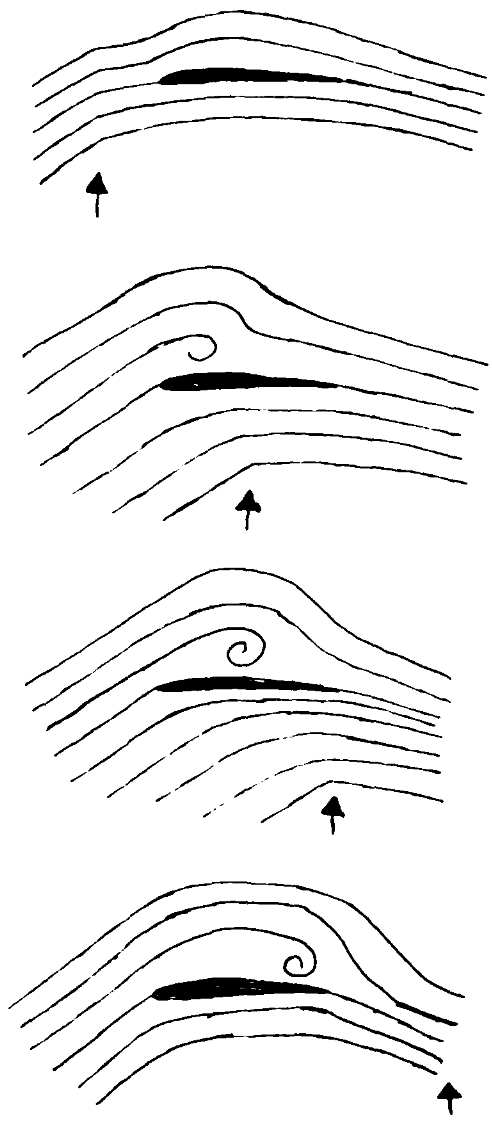


Fig.9 Available manoeuvre acceleration if given value of lift coefficient is not to be exceeded on encountering a 15 m/sec gust



position of gust front

Fig.10 Sketch of flow over a wing stalled by encounter with a sharp edged gust

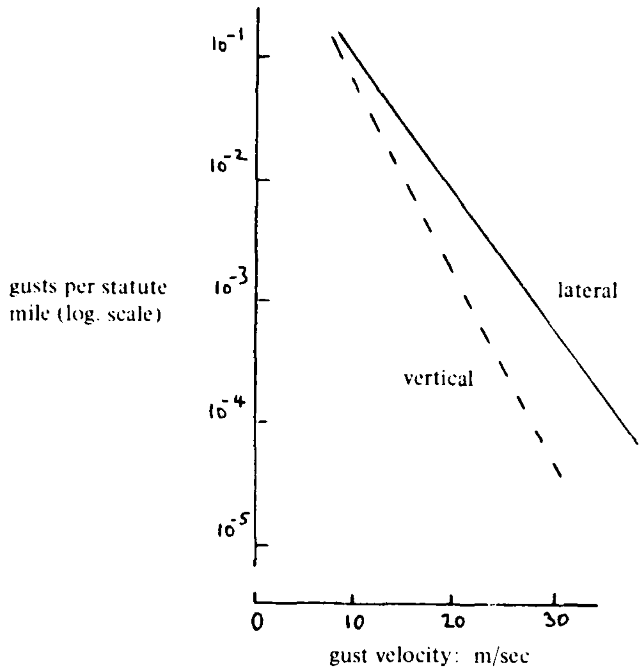


Fig.11 Turbulence data from Ref.2 used in present analysis

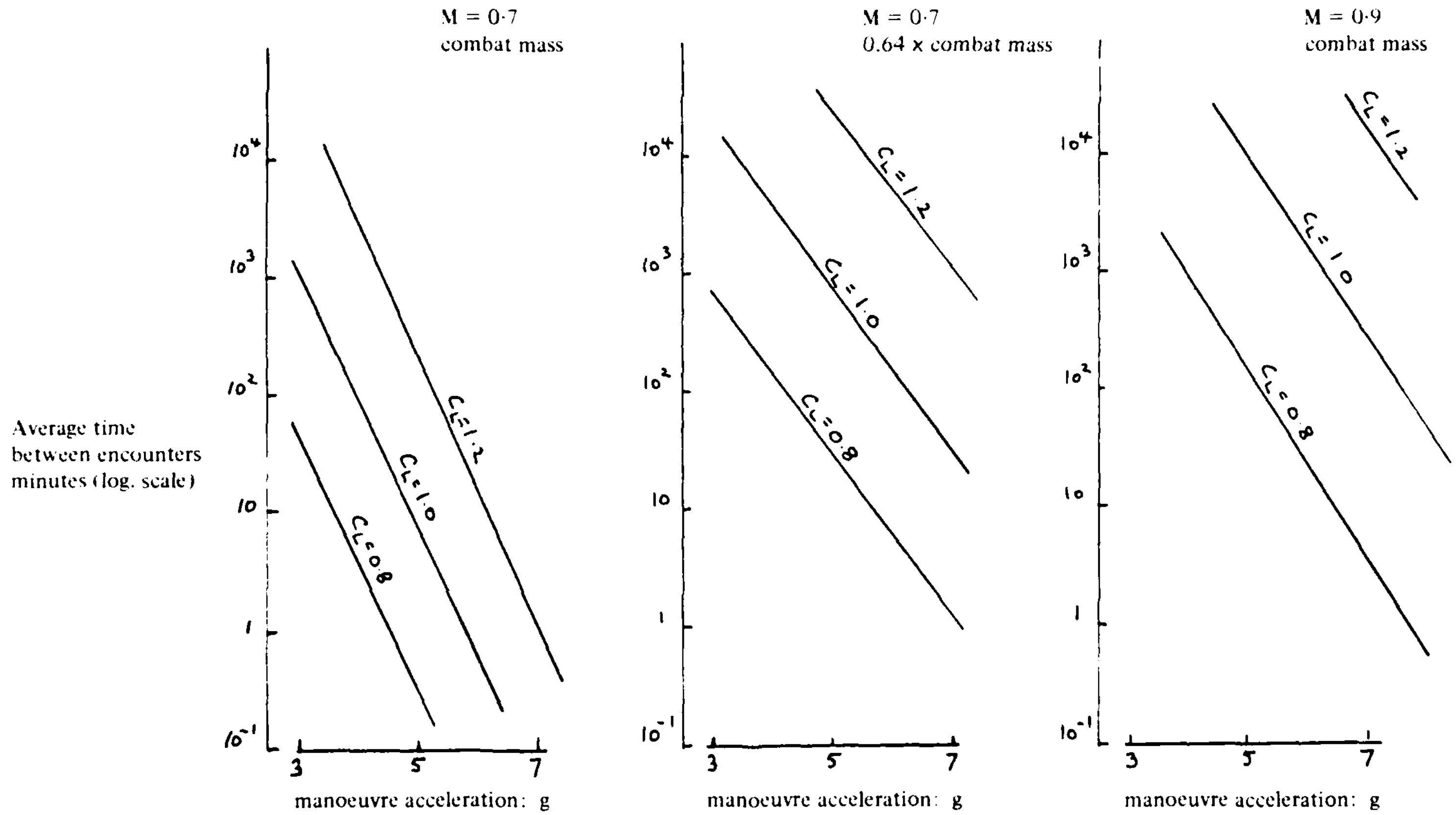


Fig.12 Average line between encounters with given values of lift coefficient (C_L) for various values of manoeuvre acceleration using turbulence data from Ref.2

ARMY HELICOPTER CRASHWORTHINESS

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SUMMARY

Although significant strides have been made in recent years toward improving aviation safety, mishaps involving all classes of helicopters presently are and will continue to be a major, expensive US Army problem in terms of casualties, materiel loss, and reduction in mission effectiveness. Modern-day training and tactical employment requirements for the US Army helicopter dictate that a large percentage of operations occur in the low-speed, low-altitude flight regime, which contributes to the problem by reducing critical margins of safety normally associated with higher airspeed and higher altitude operations with accompanying greater time for response in case of an emergency. This increased probability of accident occurrence, coupled with the lack of an in-flight egress capability, makes design for crashworthiness essential for Army helicopters.

This paper discusses the evolution of crash survival design criteria, its influence on the formulation of a US Army military standard for rotary-wing aircraft crashworthiness, and its application to current and new-generation Army helicopters. Emphasis is given to the need for a total systems' approach in design for crashworthiness and the necessity for considering crashworthiness early in the design phase of a new aviation weapon systems development effort. The actual application of crashworthiness to Army helicopters is presented with statistics that show dramatic reductions in fatalities and injuries with implementation of a crashworthy fuel system. Current and planned US Army R&D to improve crashworthiness technology is discussed, including full-scale crash testing, human tolerance definition, improved energy absorbers, crew restraint systems, and crash impact characteristics of composite helicopter structures. Applicability of the work within Army helicopter crashworthiness to commercial/civil helicopters is shown. The cost effective aspects of designing helicopters to be more crash survivable are also discussed.

INTRODUCTION

Research investigations directed toward improving occupant survival and reducing materiel losses in aircraft crashes have been conducted by the Army for more than 20 years. However, up until approximately 10 years ago the principal emphasis within Army aviation survivability was placed on accident prevention. Although this is indeed the ultimate objective deserving priority effort, past experience clearly shows that accident prevention alone simply is not sufficient. Mishaps of all natures involving Army aircraft have been, are, and will continue to be a major, expensive problem. Research has been accomplished on accidents worldwide involving Army aviation, and accident histories are routinely disseminated throughout the Army. Unfortunately, too many lessons learned from these accident histories are not applied and hazardous design features, human errors, and operational errors are repeated year after year. Too many Army aircrewmen are still being fatally injured in potentially survivable accidents, and the percentage of major injuries and rate of materiel losses are still unacceptably high. There is no easy solution to the problem. Significant gains can be made, however, toward reducing these unacceptable accident losses, but to do so we must aggressively pursue a program that addresses key issues of both accident prevention and crashworthiness design. Since the helicopter's potential for accident is great due to its mission and the environment in which it must accomplish that mission, it is imperative that it be engineered to minimize damage and enhance occupant survival in crashes. In designing helicopters to be more crash survivable, two subissues then become paramount: establishing viable crashworthiness design criteria, and the more difficult task, applying these crashworthiness criteria to Army aircraft design.

To help establish the severity of the problem within US Army aviation, Table 1 provides a summary of accident statistics for Army helicopters for the period of time from 1972 to 1982. With the exception of the OH-6, these aircraft are still in the operational fleet and comprise the bulk of the Army's helicopters. None of these aircraft had crashworthiness in their basic design. During the period reviewed there were over 900 helicopter incidents/accidents with over 400 occupant fatalities. The fatalities would, without question, have been more severe had not the aircraft been retrofitted in the early to mid-70s with crashworthy fuel systems. Considering the personnel aspects in the crashes of these helicopters, the two columns on the right reflect that there were survivors in more than 85 percent of all of the accidents, but, more important, that nearly one-third of all the fatalities occurred in accidents where there were survivors. It can be seen that the costs associated with these accidents in terms of men and materiel replacement are quite high. These costs, however, do not reflect the potentially significantly greater costs

TABLE 1. ARMY HELICOPTER ACCIDENT HISTORY 1972-1982, NONCRASHWORTHY AIRCRAFT

ACFT	NO. OF ACCID	NO. OF PSNL	NO. OF INJURIES MINOR	NO. OF INJURIES MAJOR	NO. OF FATAL-ITIES	COST-$\overline{M}$ (MEN & MATL)	% ACCID WITH SURVIVORS	% FATAL IN ACCID WITH SURVIVORS
UH-1	426	1852	300	210	229	132	82	34
OH-58	235	533	97	76	59	28	86	25
AH-1	156	301	51	37	35	52	88	26
OH-6	72	160	44	19	9	5	93	22
CH-47	32	277	36	9	100	44	78	10
TOTAL ACFT	921	3123	528	351	432	261	85	26

that are associated with loss of mission capability. Further, these statistics are based on current peacetime experience which reflects a total cumulative flight time of approximately 1½ million hours per year for Army aviation with a fatality rate of approximately 2.5 per 100,000 hours of flying time. The severity of the problem increases severalfold during periods of combat, as demonstrated in Vietnam when, during the height of the conflict, total helicopter flight time was in excess of 5 million hours per year with the fatality rate of 10 per 100,000 hours.

Data from these accident and crash injury investigations (Reference 1) have revealed deficiencies in the crashworthiness of the older, existing Army helicopters. Key deficiencies include:

- Structural collapse (roof downward and floor upward) causing loss of occupiable volume
- Inward buckling of frames, longerons, etc., causing penetration wounds to personnel
- Lethal internal structure causing head, chest and extremity injuries from occupant flailing
- Floor breakup permitting seats to tear out and occupants to become flying missiles
- Landing gear penetration into occupied areas and fuel systems causing contact injuries and fires
- Landing gears not designed for sufficiently high sink rates and insufficient deformable airframe structure permitting excessive acceleration (G) forces to be transmitted to the occupants and causing excessive materiel damage
- Intrusion of the occupied area by the main rotor gearbox and other high mass items causing crushing and contact injuries to the occupants
- Insufficient structural stiffness permitting inward crushing and entrapment of occupants in rollover accidents

It has been demonstrated, however, that significant gains can be made toward reducing the severity of these and related problems through the judicious development and application of crashworthiness design features into Army aircraft.

CRASHWORTHINESS DESIGN CRITERIA

In-depth assessment of available crash data was first accomplished in the mid-60s by a joint Government/industry review team. The product of that team was the world's first crash survival design guide for light fixed- and rotary-wing aircraft, published in 1967. Revisions to this guide were made in 1969, 1971, and 1980 (Reference 2). This design guide was subsequently converted into a military standard (MIL-STD-1290) in 1974 (Reference 3) which is presently undergoing revision. MIL-STD-1290 addresses five key areas that must be considered in designing a helicopter to conserve materiel and provide the necessary occupant protection in a crash:

- Crashworthiness of the structure--assuring that the structure has the proper strength and stiffness to maintain a livable volume for the occupants and prevent the seat attachments from breaking free
- Tie-down chain strength--assuring that the high mass items such as the transmission and engine do not break free from their mounts and penetrate occupied areas
- Occupant acceleration environment--providing the necessary crash load absorption by using crushable structures, load limiting landing gears, energy-absorbing seats, etc., to keep the loads on the occupants within human tolerance levels
- Occupant environment hazards--providing the necessary restraint systems, padding, etc., to prevent injury caused by occupant flailing
- Postcrash hazards--after the crash sequence has ended, providing protection against flammable fluid systems and permitting egress under all conditions

A survivable crash is generally defined as one wherein the impact conditions inclusive of pulse rate onset, magnitude, direction and duration of the accelerative forces that are

transmitted to the occupant do not exceed the limits of human tolerance for survival, and in which the surrounding structure remains sufficiently intact during and after impact to permit survival. Thus, helicopters designed to meet MIL-STD-1290 shall be designed to prevent occupant fatalities and minimize the number and severity of injuries during crash impacts of a severity to and including the 95th percentile potentially survivable accident while minimizing aircraft damage to the maximum extent practical. The 95th percentile design pulse generally means that the loads on the occupants would be greater in only 5 percent of the accidents but would still be within the defined human tolerance limits. Table 2 presents the 95th percentile potentially survivable crash design pulse for helicopters expressed in terms of impact velocity change with associated minimum attitude requirements. It should be noted that Table 2 and some of the subsequent discussion reflect criteria that are proposed for the revised MIL-STD-1290 and in some cases deviate from available published design criteria. This approach is meant to enhance the validity and usefulness of this paper. Perhaps the most critical MIL-STD-1290 factor in designing the helicopter for crash survivability is the vertical design impact velocity change requirement. Since the helicopter spends a large percentage of its operational life in

TABLE 2. 95TH PERCENTILE POTENTIALLY SURVIVABLE CRASH IMPACT DESIGN CONDITIONS

IMPACT DIRECTION (AIRCRAFT AXES)	OBJECT IMPACTED	VELOCITY CHANGE (FT/SEC)	MIL-STD-1290 CURRENT			MIL-STD-1290 PROPOSED REVISION		
			PITCH	ROLL	YAW	PITCH	ROLL	YAW
Longitudinal (Cockpit)	Rigid Abutment or Wall	15						
Longitudinal (Cabin)	Rigid Abutment or Wall	40						
Longitudinal (Cockpit & Cabin)	Rigid Surface	50	0	0	0	-5°	$+10^{\circ}$	0
Vertical	Rigid Surface	42	$+15^{\circ}$	$+30^{\circ}$	0	$+15^{\circ}$ to -5°	$\pm 10^{\circ}$	0
Lateral[a]	Rigid Surface	25						
Lateral[b]	Rigid Surface	30						
Resultant Vector*	Rigid Surface	50						

(a) Light fixed-wing aircraft, attack and cargo helicopters.
(b) Other helicopters.
*Note: The downward, sideward, and forward velocity components of the resultant velocity vector do not exceed 42, 30, and 50 ft/sec, respectively.

the low-speed, low-altitude flight regime, accidents predominantly occur with high vertical descent rates and with the aircraft in a near normal attitude. Thus, the aircraft must withstand vertical impacts of 42 ft/sec, within the aircraft attitude limits of ± 10 degrees roll and +15 degrees to -5 degrees pitch, (1) with no more than a 15-percent reduction in the height of the cockpit and passenger/troop compartments and (2) without causing the occupants to experience injurious accelerative loading. This is a good example of a MIL-STD-1290 proposed revision. The current version states the 42-ft/sec requirement for an aircraft impact attitude within +15 degrees pitch and +30 degrees roll, which not only dictates excessive landing gear capability but does not represent the typical impact as derived from recent analysis of accident data. Based on this recent analysis of roll and pitch frequencies for Army helicopter accidents over the past 10 years, a more detailed representation of the vertical crash impact conditions has been developed and is being proposed as a MIL-STD-1290 revision (see Figure 1). In this case pitch and roll envelopes are specified for vertical velocities of both 42 and 36 ft/sec.

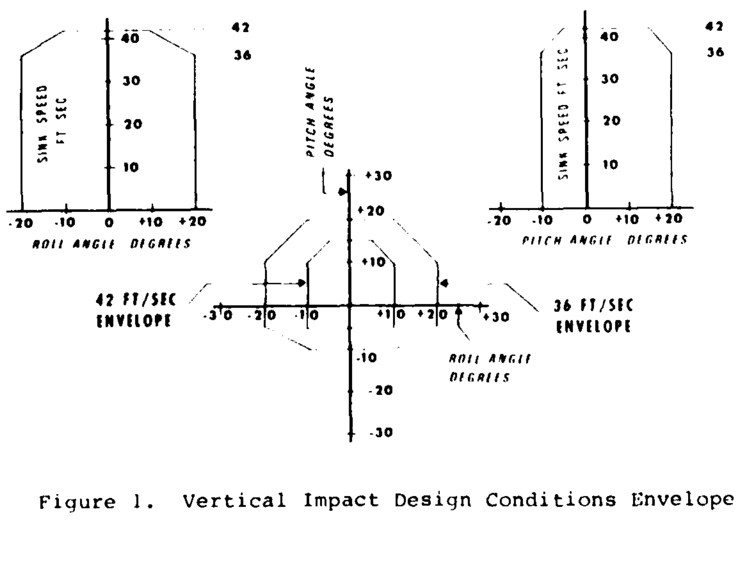

Figure 1. Vertical Impact Design Conditions Envelope.

Other key design impact velocity changes are shown in Table 3. The landing gear shall provide energy absorption capability to reduce the vertical velocity of the fuselage as much as possible under the crash conditions. As a minimum, the landing gear shall be capable of decelerating the aircraft at normal gross weight from an impact velocity of 20 ft/sec onto a level rigid surface within an attitude envelope of ± 10 degrees roll and +15 degrees to -5 degrees pitch without allowing the fuselage to contact the ground and without gear penetration into an occupied area. Plastic deformation of the landing gear and its mounting system is acceptable in meeting this

requirement; however, with the possible exception of the rotor blades, the remainder of the aircraft structure shall be flightworthy after impact. The nose section of the helicopter is to be designed to preclude earth plowing and scooping tendencies when the forward 25 percent of the fuselage is subjected to a longitudinal load uniformly applied with a local upward load of 10 Gs and a rearward load of 4 Gs. The fuselage shall also be designed for rollover protection and shall be capable of sustaining a 4 G load applied uniformly over the fuselage to surface contact area if rollover occurs. Finally, all high mass items which would pose a hazard to personnel during a crash shall be designed to withstand 20 Gs longitudinally, 20 Gs vertically, and 18 Gs laterally when applied separately.

TABLE 3. ADDITIONAL MIL-STD-1290 DESIGN REQUIREMENTS

Landing Gear	20 Ft/Sec, $\pm$ 10° Roll, +15° to -5° Pitch No Fuselage Damage
Plowing & Scooping	10 G Up & 4 G Aft on Fwd 25% Fuselage
Rollover	4 G Side Load 4 G Roof Load
High Mass Tie-down (Applied Separately)	$\pm$ 20 G Longitudinal + 20 G/-10 G Vertical $\pm$ 18 G Lateral

For maximum effectiveness, design for crashworthiness dictates that a total systems approach be used and that the designer consider survivability issues in the same light as other key design considerations such as weight, load factor, and fatigue life during the initial design phase of the helicopter. Figure 2 depicts the system's approach required relative to management of the crash energy for occupant survival for the 95th percentile vertical crash pulse design condition. The crash G loads must be brought to within human tolerance limits in a controlled manner to prevent injury to the occupants; this can be accomplished by using the landing gear, floor structure, and seat to progressively absorb most of the crash energy during the crash sequence. Thus, the occupant is slowed down in a controlled manner by stroking/failing the landing gear, crushing the floor structure, and stroking the seat at a predetermined load before being subjected to the crash pulse which by then has been reduced to within human tolerance limits. In addition, the large mass items such as the overhead gearbox are slowed down by stroking/failing of the landing gear or fuselage structure, and in some cases, by stroking of the gearbox within its mounts. In this example, assuming that the landing gear has been designed to meet the minimum requirements of MIL-STD-1290, i.e., 20 ft/sec, the fuselage would be decelerated to approximately 37 ft/sec at the time of contact with the surface.

The Army's most recent helicopters, the UH-60 BLACK HAWK and AH-64 APACHE, are both designed generally in accordance with the requirements of MIL-STD-1290, and the significant payoff for designing these aircraft for crashworthiness will be addressed later.

The preceding discussion should not be interpreted, however, to imply that nothing can be done for existing aircraft systems. Quite the contrary. Considerable improvement in crashworthiness can be achieved on existing helicopters by applying such features as improved crew restraint systems, energy-absorbing seats, crash tolerant fuel systems, and breakaway control sticks.

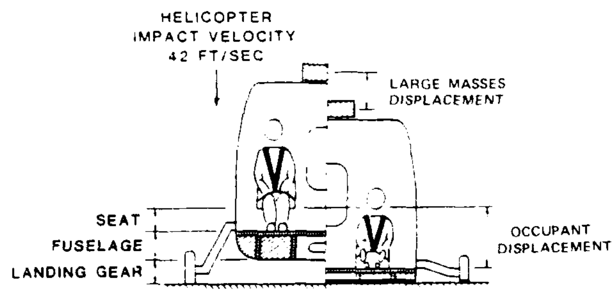

Figure 2. Energy Management System.

Also, the above discussion of crashworthy requirements principally addresses the airframe, the main objective of which is to provide a protective shell for the occupants and to allow deformation of the structure in a controlled, predictable manner to minimize forces on the occupants. Other key requirements in MIL-STD-1290 in designing a helicopter system for crashworthiness include:

- Occupant restraint design--Seats and litters shall be designed to retain occupant position during crash and shall contain integral means of crash force attenuation. Crew seats shall be designed to permit the seat to stroke 12 inches vertically. The immediate surroundings shall be designed to minimize occupant injury when body parts flail in a crash. These designs shall be applicable with the 5th through the 95th percentile male aircrewman (i.e., 133-lb thru 212-lb crewman).

- Cargo and equipment restraint system design--The design shall provide sufficient restraint of all cargo and high mass equipment in all directions to prevent injury to occupants in the 95th percentile survivable accident.

- Postcrash fire prevention design--All flammable fluid systems shall be designed to minimize spillage of fluids during and after survivable crash impacts, and when spillage cannot be avoided, the system shall be designed to prevent ignition of the fluids to the maximum extent practical.

- Postcrash emergency escape provisions design--The design shall provide for sufficient size and quantity of exits to allow occupant escape within 30 seconds after the crash sequence is over (including ditching) even when half of the exits are blocked off.

CRASHWORTHY R&D PROGRAM HIGHLIGHTS

Considerable effort has been accomplished in the past and is presently ongoing in the area of helicopter crashworthiness research and development. Efforts include such diverse areas as human tolerance definition, crashworthy troop and crew seats, improved restraint systems, crashworthy fuel systems, math modeling of crashworthy structures, crashworthy composite structures, and full-scale crash testing of both crashworthy and noncrashworthy aircraft. Results of these efforts are applicable to both the retrofit of existing aircraft systems, to improve their survivability/mission capability, and to the definition of design criteria and publication of specifications and standards for crashworthy design of new systems. Highlights of key crashworthy R&D programs within the US Army are presented in the following paragraphs.

Crash Impact Characteristics of Helicopter Composite Structures

In recent years, composite materials such as graphite, fiberglass, boron and Kevlar have been used more extensively in the design of structural and nonstructural aircraft components due to their potential for cost and weight savings. Entire composite airframes have already been produced for general aviation fixed-wing aircraft. It is therefore reasonable to assume that in the near future the helicopter industry will be producing large numbers of aircraft with major structural components, such as the fuselage, wings, empennage, blades, and landing gear, constructed of composite materials. In view of the crashworthiness requirements specified in *MIL-STD-1290*, it was considered particularly appropriate during the early stage of application of composite materials to helicopter structures to investigate their crash impact behavior. When applying composites to a crashworthy airframe structure, entirely different design concepts may be required than are used with conventional metal structures. Composite materials exhibit low strain-to-failure compared to such metals as 2024 aluminum, a ductile metal that can tolerate rather large strains, deform plastically, and absorb a considerable amount of energy without fracture. Because of this difference between composites and metals, crash energy absorption with composites will not come through material stress-strain behavior as it did with metals, but rather through innovative design.

To determine the crashworthiness characteristics of helicopter composite structures, a program was conducted for the design, fabrication, design support testing, analysis and crash testing of two full-scale composite helicopter cabin sections. Crash tests were conducted for 0- and 20-degree roll impact attitudes at a vertical impact velocity of 30 ft/sec, which is representative of the Army's vertical impact velocity requirement (MIL-STD-1290) for noninjurious loadings if the landing gear is assumed to absorb approximately one-half the impact energy.

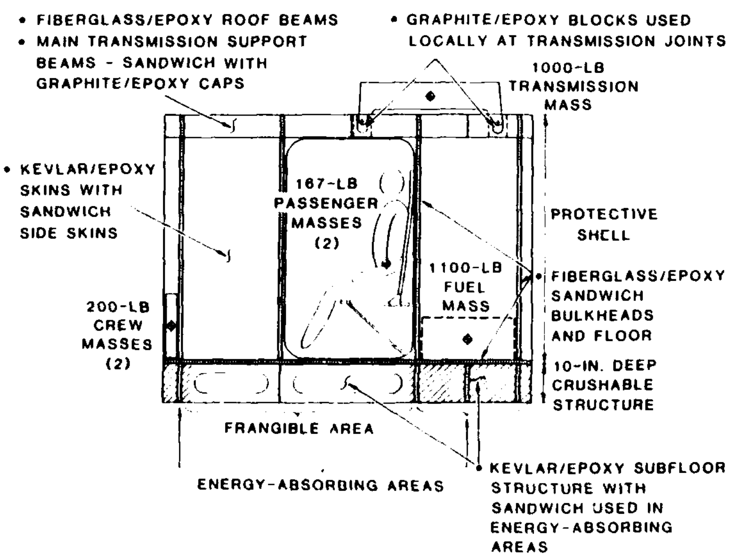

Figure 3. Composite Cabin Test Section Design Features.

The cabin section (Figure 3) had three major bulkheads: the forward crew bulkhead, the aft cabin bulkhead, and the aft fuel cell bulkhead. The masses in the cabin were a 1000-pound overhead transmission mass, 1100 pounds of fuel, two 167-pound passenger masses, and two 200-pound forward crew masses. Located directly beneath the forward crew bulkhead and the aft two bulkheads was a crushable cabin subfloor structure designed to absorb energy and control loads to the primary protective shell. Between the crush zones was frangible structure that would crush out of the way without damaging the floor structure.

Kevlar, fiberglass, and graphite/epoxy materials were used in constructing the cabin section. Fiberglass sandwich construction was used in the bulkheads and floor panel, and in portions of the upper roof structure. Kevlar was used in the roof, belly and side skins, and in the energy-absorbing subfloor structure. Graphite reinforcement was used around the door frame and in the main roof beams. Figure 4 depicts the cabin components. For the first test, two stroking seats equipped with wire-roller attenuators were installed in the cabin. The right seat was floor-mounted and the left seat was bulkhead-mounted. Fiftieth percentile Part 572, Hybrid II anthropomorphic testing dummies were placed in the seats. The impact was on a simulated rigid surface comprised of steel plates over a sand base. Instrumentation included accelerometers on major masses and important structural locations, and high-speed motion picture cameras were used to record the structure response and failure modes. The level attitude 30 ft/sec vertical velocity (or about 14 feet free fall height) composite cabin section drop test was survivable based on the excellent post-test condition of the cabin protective shell structure and the performance of the energy-absorbing structure components (see Figure 5). There was approximately 4 inches of subfloor crush. The bulkhead-mounted left seat stroked 9 inches and the floor-mounted right

seat stroked 6 inches. Figure 6 depicts the vertical acceleration time history of the left troop seat dummy pelvis and floor for the flat impact test. This vividly shows how the stroking seat prevented injurious accelerative loadings from being transmitted to the dummy. Both seats had an 11-inch stroke capability and therefore did not bottom out. The 1000-pound overhead transmission mass stroked about 1.25 inches in the specially designed roof beam attachment joints.

The second cabin section was equipped with lumped masses for the passengers in lieu of stroking seats and dummies. This was done in order to reduce the complexity of the test specimen in the severe 20-degree roll impact condition. During the test, the cabin impacted on the left side at 20-degrees roll with very little rotation before the vertical velocity was attenuated. The cabin then rolled over on the right side with little vertical kinetic energy at that time, which is indicated by the right side crushable structure not being damaged (see Figure 7). The test verified that the crushable energy-absorbing structure can tolerate an oblique impact with combined loading and still perform well and protect the structure surrounding the occupied volume.

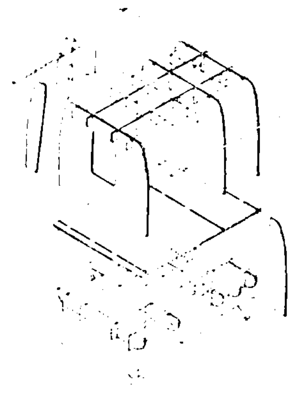

Figure 4. Composite Cabin Components.

An important part of this program was to evaluate analysis methods that could be useful tools in future design of crashworthy structures. The KRASH and DYCAST computer programs were used for dynamic analysis of the crash impact conditions, while NASTRAN was used to develop internal loads in the structure. Load factors were determined from the

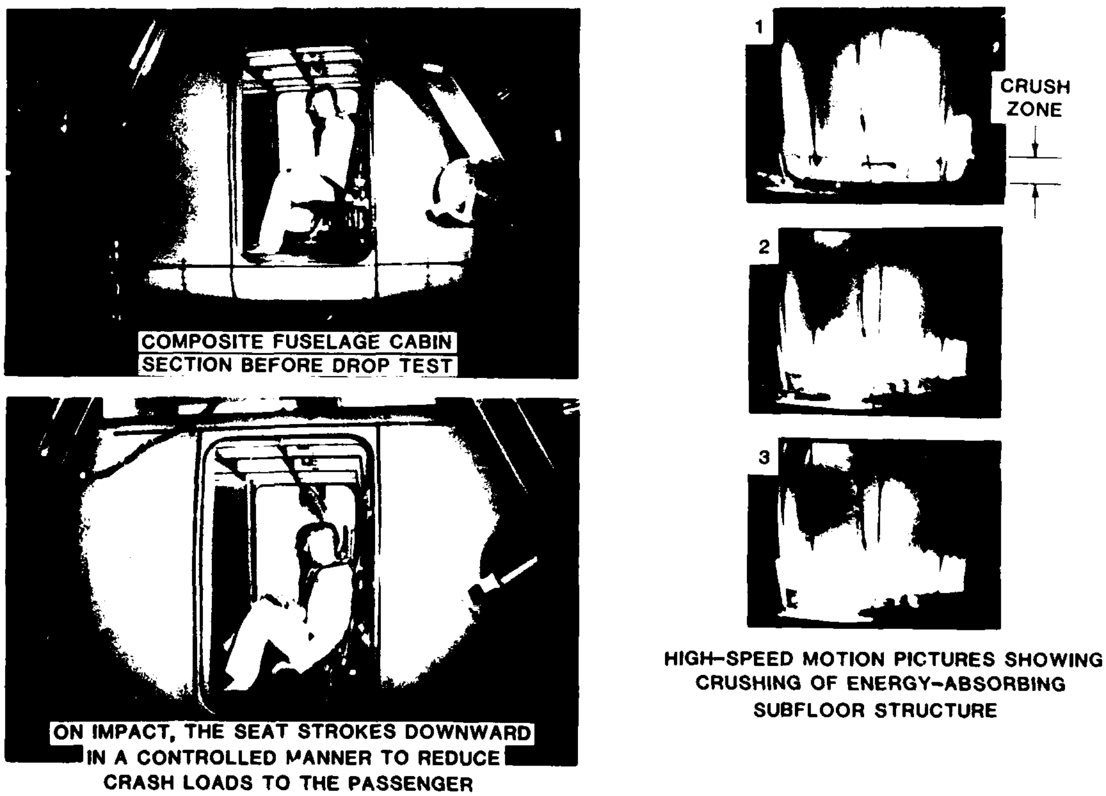

Figure 5. Flat Drop Test.

KRASH dynamic analysis for major mass items such as the crew, troops, fuel, and transmission and were applied to the NASTRAN finite element model of the cabin section. In addition, the crush zone loads were applied to the floor. The NASTRAN model was then used to develop internal loads to be used for the strength analysis. Critical areas in the primary structure components were sized using the internal loads from the NASTRAN mode. Some of the important critical areas were the main roof beams that support the transmission mass, the aft cabin bulkhead, the roof and side skins, and the floor panel loaded by the fuel mass. The crushable subfloor structure was sized based on design support testing data to get the proper energy absorption and control of loads to the primary protective shell structure. As a result of this research, it was concluded that:

- MIL-STD-1290 crashworthiness requirements can be met with a composite fuselage structure if designed with energy absorption and load attenuation in controlled areas.

- The use of design support testing to size and optimize the load deflection characteristics of composite material energy-absorbing components is an accurate, economical approach.
- The KRASH (supplemented with NASTRAN) and the DYCAST nonlinear, large-deflection structure crash simulation computer programs can be useful and reasonably accurate analytical tools for designing the crashworthy composite fuselage cabin sections.

Additional details on this program are provided in Reference 4.

Human Tolerance

A major objective of Army crashworthiness is to attenuate crash loads reaching the occupants to levels within the limits of human tolerance. To properly design to meet this objective, limits of human tolerance to acceleration about all aircraft axes must be accurately defined. This is an extremely difficult set of data to obtain since human tolerance to impact forces varies appreciably with an individual's age, sex, weight distribution, and general state of health. Army helicopters can normally be expected to be occupied by personnel who are younger and in better physical condition than that of the general population for which most of the tolerance data have been developed to date. Thus, a degree of conservatism may be built in for the military in using criteria developed from a "general public" cross section. However, these tolerance criteria have for the most part been based on experiments involving subjects seated with a "correct" upright posture, while Army aviators spend large portions of their time while in the aircraft in less-than-ideal postures for absorbing crash impact (e.g., viewing through target designating/sighting scopes). During nap-of-the-earth flight operations, a crewman can expect little warning of an impending crash impact and will virtually have no time for assuming a proper pre-impact posture.

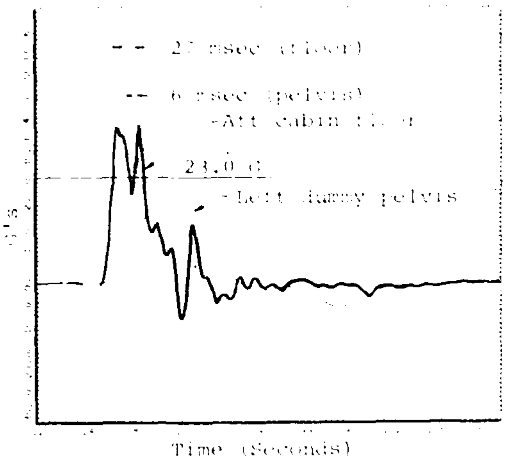

Figure 6. Vertical Acceleration Time-History of Left Troop Seat Dummy Pelvis and Floor For Flat Impact.

With proper restraint, aircraft occupants can withstand the full 95th percentile survivable crash acceleration conditions in the lateral (G_y) and longitudinal (G_x) directions with no energy absorption. Such is not the case with accelerations directed along the vertical axis, particularly headward ($+G_z$) acceleration. The lumbar vertebrae of the occupant, which must support most of the upper torso loaded as a column, are susceptible to compression fracture with attendant injuries such as paralysis. To prevent the occupant from experiencing injurious accelerative loadings, energy attenuation, in the form of energy-absorbing landing gear, crushable belly structure, and stroking seats, is required to control vertical loads.

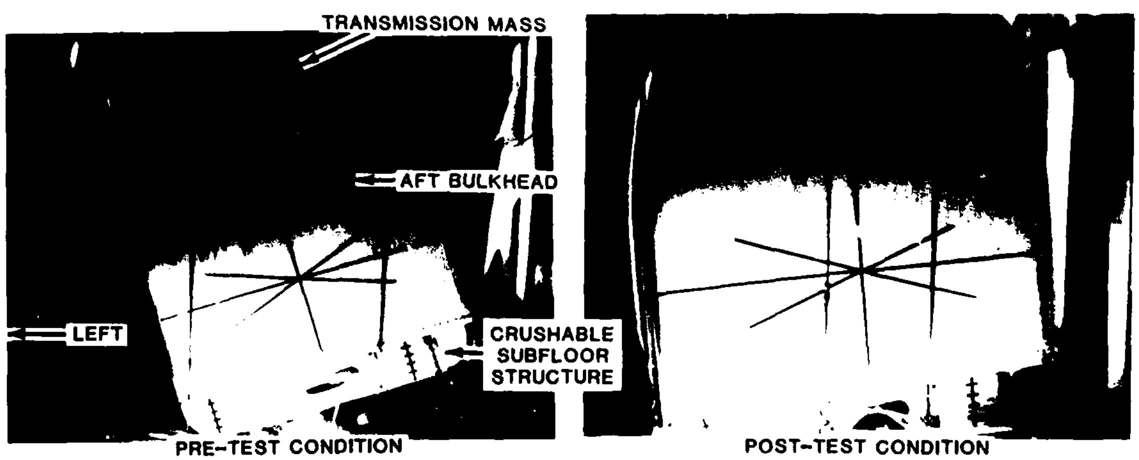

Figure 7. Twenty-Degree Roll Drop Test.

Current Army criteria are based on the Eiband[7] human tolerance data for the upper limit of tolerable (with no injury) acceleration in the $+G_z$ direction (see Figure 8). These data establish the upper limit for vertical acceleration excursions transmitted to the occupant to magnitudes of less than 23 G for time durations exceeding 6 milliseconds. The Army crashworthy crewseat specification, MIL-S-58095[6], has placed this limitation on the seat pan accelerations while the seat is subjected to dynamic testing defined in Figure 9. In seat tests conducted since the specification was established in 1971, a characteristic curve (Figure 10) shows that the seat pan deceleration rises sharply during the onset of the input pulse, then drops rapidly as the seat becomes fully coupled to the

occupant. The deceleration may actually pass through zero (constant velocity) as this event occurs. The deceleration then rises sharply and forms a secondary spike before damping out around the load factor used in the design of the seat energy-absorbing system. The primary seat pan deceleration spike is of little concern since it represents the response of the unloaded seat to the impact event. The secondary spike, however, occurs after the seat cushion and buttocks have compressed and, in most tests, its duration above 23 G exceeds the Eiband injury criteria. The body is a complex dynamic system when one considers the pelvis, chest and head as masses being interconnected by flesh and spinal column "springs." Whether the characteristic secondary spike indeed applies injurious loads to any part of the spine is still largely unknown. Accordingly, research is being conducted to better define human tolerance to injury as related to the typical Army aviator. This includes advanced energy absorber research and research involving cadaver testing.

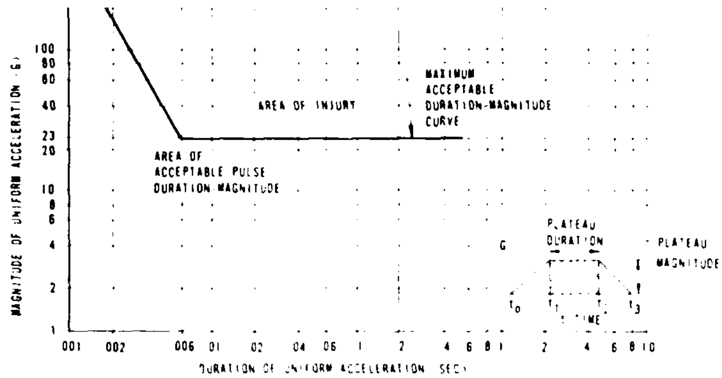

Figure 8. Maximum Acceptable Vertical Pulse Acceleration and Duration Values.

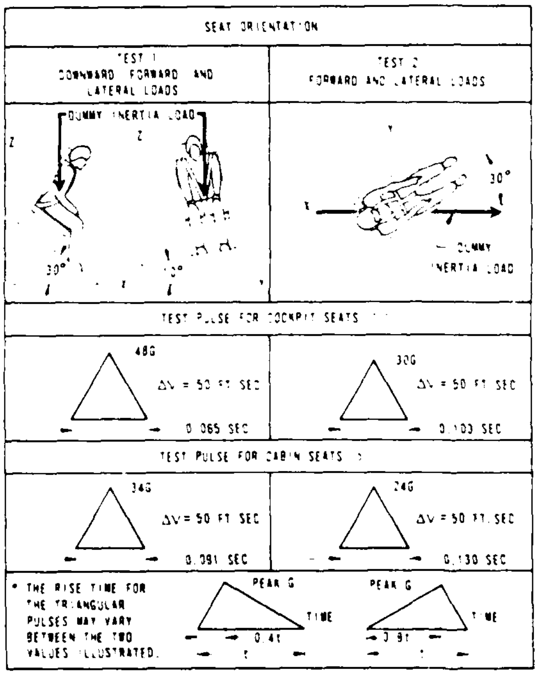

Figure 9. Dynamic Test Conditions for Aircraft Seats.

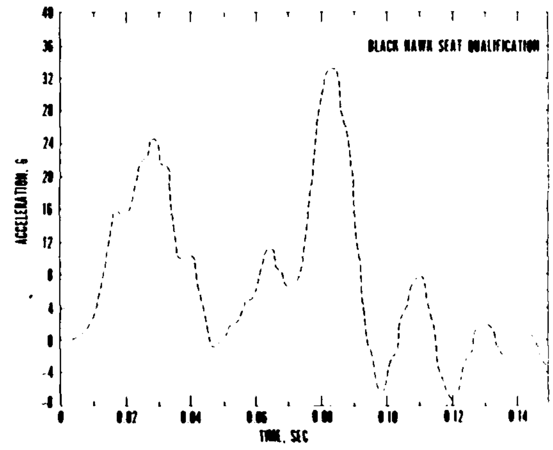

Figure 10. MIL-S-58095 Test No. 1 Seat Pan Vertical Accelerations.

Since mid-1979, the Army has been jointly involved with its sister services and the FAA in sponsoring tests to establish the threshold of human tolerance to spinal compressive loads. Testing has been performed at the Wayne State University Bioengineering Center on their WHAM III (Wayne Horizontal Accelerator Mechanism) sled. This testing has involved the use of human cadavers in both rigid (nonstroking) seats and the production BLACK HAWK helicopter crashworthy crewseat. Tests of three embalmed cadavers in the rigid seat gave mixed results, with spinal fractures occurring at 7.5 G, 28.5 G, and 13.0 G. These results were achieved by testing each cadaver to progressively higher peak impact G loading with the impact vector being parallel to his spine. X-rays were taken between runs until a spinal fracture was indicated. Table 4 summarizes the results for these three tests.

An unembalmed cadaver test series is presently ongoing using the BLACK HAWK crewseat having a 12- to 17-inch stroking capability depending on seat height adjustment. This testing has been performed with two seat orientations: one to simulate a "flat" or 0-degree pitch angle BLACK HAWK ground impact (referred to as the "vertical" mode) and another to simulate a 30-degree nose-down BLACK HAWK ground impact (referred to as the "combined axis" mode). The sled impact pulse has approximated a 41 G triangular pulse of 64 milliseconds duration for a velocity change of 42 ft sec. This is representative of the Army's 95th percentile potentially survivable impact. Testing has been conducted

TABLE 4. SUMMARY OF TEST CONDITIONS FOR RIGID SEAT TESTS WITH EMBALMED CADAVERS

TEST NO.	CADAVER NO.	AGE	HEIGHT	WEIGHT (LB)	SEX	PEAK ACCEL. (G)	FRACTURE CONDITION
SERIES #1 (3 RUNS)	4612	52	5' 10"	161	M	4,6,8	T9 @ 7.5 G
SERIES #2 (11 RUNS)	4654	49	5' 7"	202	M	4 TO 30	T10 & T11 @ 28.5 G, COMPRESSION FAILURE
SERIES #3 (8 RUNS)	4660	51	5' 7"	216	M	4 TO 30	T8 @ 13 G, ANTERIOR WEDGE FRACTURE

with seat energy attenuators (EA's) set for 14.5, 11.5, and 8.5 G stroking loads based on a 50th percentile seat occupant. Table 5 lists pertinent data relating to each test in this series with cadaver injury condition determined by post-test autopsy.

TABLE 5. SUMMARY OF TEST CONDITIONS FOR UH-60A CREWSEAT TESTS WITH UNEMBALMED CADAVERS

TEST NO.	TEST CONDITION	IMPACT MODE	AGE SEX	HEIGHT	WEIGHT (LB)	INPUT VELOCITY CHANGE (FT/SEC)	INPUT PEAK ACCEL. (G)	VERTEBRAL INJURY CONDITION
AF020	14.5 G E A	VERT	44 F	5' 3"	166	41.5	43.4	NONE
AF021	14.5 G E A	COMB	44 F	5' 3"	166	42.6	44.4	T12 END PLATE, C1-C2 ARTICULATION
AF025	14.5 G E A	VERT	55 M	5' 7"	160	-	-	L3, ANTERIOR WEDGE FRACTURE
AF028	14.5 G E A	VERT	61 F	5' 4"	140	45.5	43.2	NONE
AF029	14.5 G E A	COMB	61 F	5' 4"	140	44.0	42.4	T12, ANTERIOR WEDGE FRACTURE
AF031	14.5 G E A	VERT	63 F	5' 5½"	148	43.0	39.8	T8, COMPRESSION FRACTURE
AF033	11.5 G E A	COMB	52 M	5' 9"	218	41.8	45.0	L1, ANTERIOR WEDGE FRACTURE
AF035	11.5 G E A	COMB	63 M	5' 8"	141	44.7	40.5	L3, ANTERIOR WEDGE FRACTURE
AF037	11.5 G E A	COMB	58 F	5' 3"	160	46.6	40.9	L3, ANTERIOR WEDGE FRACTURE
AF039	8.5 G E A	COMB	52 M	5' 10"	200	41.0	37.9	NONE
AF040	8.5 G E A	COMB	63 M	5' 6½"	142	38.0	35.7	L2, END PLATE FRACTURE
AF041	8.5 G E A	COMB	[illegible] M	5' 10"	165	36.2	35.0	NONE
AF042	8.5 G E A	COMB	47 M	5' 10"	155	42.2	42.9	C2 FRACTURE, T9, L4 COMPRESSION FRACTURE

The median age of cadavers tested to date is 54.6 years. Questions have been raised (and justly so) regarding any differences in spinal compressive strength that may exist between these cadavers and the younger occupants typically involved in Army aviation mishaps. With each control is exercised over cadaver selection by rejecting any having died from long-term or degenerative illnesses, other factors relating to aging such as osteoporosis may be present. Medical doctors associated with the program have estimated spinal tolerance of these cadavers to be approximately one-half that of Army aviators. Crash tests were performed on excised spines from six of the test cadavers to determine their stiffness and ultimate compressive strength. Bone mineral assay tests were also performed in an attempt to achieve a mineral content-to-strength correlation. Neither of these procedures yielded [illegible] results, it is felt mainly due to the low sample size.

Throughout the [illegible] testing, the dynamic behavior of the test subjects [illegible], particularly in the combined axis impacts. High speed [illegible] most subjects underwent severe hyperflexion in spite of the [illegible] pretensioning of the 5-point restraint harness built into the [illegible] lap belt and each shoulder harness are tightened to 50 pounds and [illegible]. In several cases, the head [illegible] between the knees at the peak [illegible] test run, the subject's shoulders and upper torso [illegible] [illegible] been seen previously in comparable tests with [illegible] The observed motion, if present in live occupants, [illegible] of injury: (1) the anterior "wedge-type" vertebral [illegible] on the anterior side of the vertebrae and (2) secondary [illegible] the head neck contact [illegible] cockpit furnishings [illegible]

Two additional tests are presently scheduled [illegible] simulation of helicopter crash impact [illegible]

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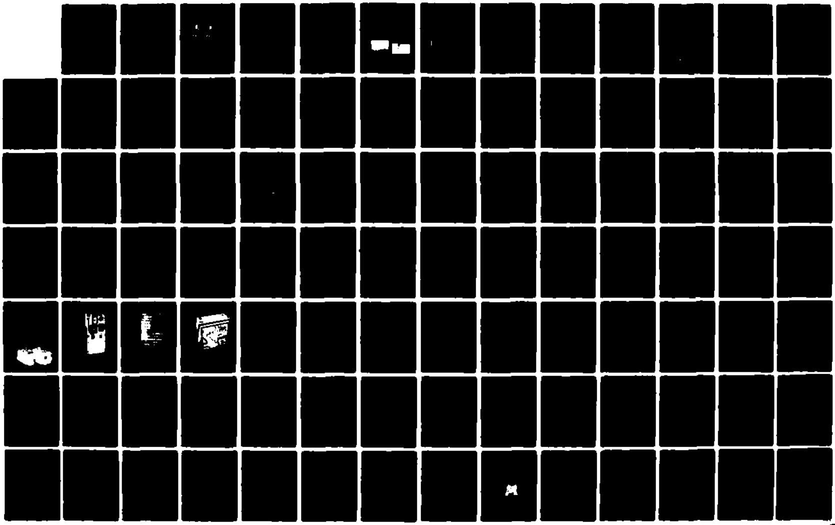

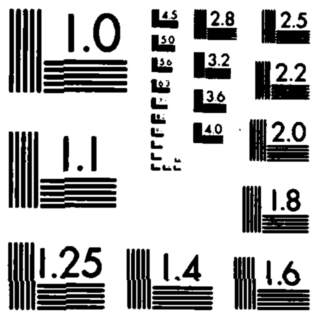

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studies, much work remains in establishing human tolerance levels from cadaveric data.

Crashworthy Seat Design Criteria

In addition to crashworthy armored crewseats in the Army's UH-60 and AH-64 helicopters, lightweight crashworthy troop seats have been developed and are installed in the cargo/troop compartment of the UH-60. These seats are constructed of tubing covered with fabric and are ceiling-suspended and floor-stabilized to provide energy attenuation in the vertical, forward and lateral directions. A compact wire-bending attenuator is used for vertical impact loads and a four-point restraint system having a single release buckle is attached to the seat. These troop seats, weighing approximately 18 pounds each, have quick disconnect fittings allowing for quick conversion of the aircraft to carry cargo.

Because of the need to develop improved criteria for the load-deflection characteristics of crashworthy seat energy absorbers, an extensive test program was initiated by the Army with joint participation by the FAA's Civil Aeromedical Institute (CAMI) and the Naval Air Development Center (NADC). A matrix of 59 dynamic impact tests were conducted using a production US Army BLACK HAWK helicopter crewseat as a baseline. Variables that were investigated included the shape, magnitude, and rate of onset of the input deceleration pulse; the velocity change; the type and size of the anthropomorphic dummy; the energy absorber limit load; the movable seat weight; the seat cushion characteristics and orientation to the input pulse; and the structural spring rate of the seat. Simula, Inc., provided test support, data reduction and analysis, and correlation of results with computer program SOM-LA (Seat/Occupant Model - Light Aircraft). Testing was conducted at CAMI (47 tests), NADC (9 tests), and Simula, Inc. (3 tests) to assess the effects on dynamic response of each test facility's unique input pulse shape. Figure 11 shows baseline 42 G input pulses produced by each test facility compared to the idealized prescribed triangular pulse.

Though space limitations do not permit the relating of results of each parametric variable, important relationships and sensitivities were established. For example, tests with "ramped" energy absorbers, whose loads increased throughout the seat stroke, revealed that they performed less efficiently than conventional square-wave type devices. The ramped devices caused the dummy to utilize more than 1.5 inches of additional stroke, while the measured accelerations and calculated dynamic response indices (DRI's) were actually higher. The dynamic response index is a dimensionless parameter resulting from a single lumped-mass, damped-spring model of the body mass acting on the human spine. It represents the human response to short-duration accelerations applied in an upward vertical direction parallel to the spine. The US Air Force uses DRI as one of its ejection seat acceptance parameters.

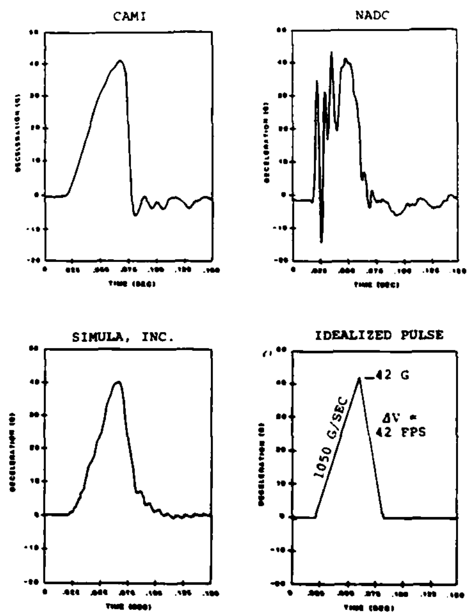

Figure 11. Typical Baseline Deceleration Pulses Compared to Idealized Pulse.

Another significant test series, sponsored by the US Army Aeromedical Research Laboratory (USAARL), contributed to the overall interpretation of data obtained from all of these test programs. A 50th percentile (Part 572) anthropomorphic dummy and a 95th percentile (VIP-95) dummy were modified to install a six-axis load cell at the base of their lumbar spines. The VIP-95 also received a six-axis cervical load cell. The dummies were then subjected to several dynamic impact tests at CAMI, some of which duplicated earlier test conditions. Results indicated that direct measurement of spinal loads in this fashion may provide a better standard for judging crashworthy seat injury criteria. More work needs to be done in order to achieve a noninterfering load cell installation, since duplicative test conditions revealed some changes in the modified dummy dynamic behavior. Reference 7 reports results under this effort and includes recommendations for future crashworthy seat specification updating.

Restraint System

The occupant restraint system is literally the "first line of defense" in preventing aircraft crash injuries. This system includes not only the belted occupant restraint but also a properly engineered mounting of the seat in the aircraft. This combination keeps the occupant from becoming a flying missile during the crash sequence. MIL-S-58095 has been the Army's crashworthy pilot/copilot seat and restraint system criteria document since 1971. A five-strap belted restraint is required consisting of the lap belts, two shoulder straps with an inertia reel, a negative G strap, and a single point of attachment buckle. The negative G strap is permanently affixed to the buckle and requires use at all times to ensure against occupant submarining under the lap belt.

The compactness of today's cockpit and the close proximity of mission equipment pose serious crash impact hazards to the aircrew. Although not desirable from a crashworthiness standpoint, operational considerations dictate that mission equipment and structure be located within the occupant's crash impact motion envelope. Given this situation, it is critical to the occupant's crash impact survival chances that he be provided with a

restraint system that minimizes his crash impact motion envelope, particularly concerning his head.

During 1979, the Army tested seven types of pilot/copilot restraint systems under dynamic impact conditions representative of various degrees of survivable crash conditions. Thirty-three dynamic impact sled tests were performed using a 95th percentile anthropomorphic dummy as the occupant. Restraints tested represented a cross section of those currently available with features such as a reflected strap shoulder harness and power haulback inertia reels. Another concept tested was a joint Army/Navy modification to the MIL-S-58095 restraint called the Inflatable Body and Head Restraint System (IBAHRS), which capitalizes on automotive air-bag technology to better restrain the occupant in severe impacts. This system (Figure 12) uses an airframe-mounted crash sensor to identify a crash condition and then trigger the inflation of air bags sewn into each shoulder harness. Inflation is accomplished by a solid propellant gas generator within approximately 25 milliseconds from being triggered, and the air bags remain inflated for approximately 1½ seconds. The inflated bags act to tighten the restraint about the crewman, better distribute the decelerative loads over his upper torso, and decrease head and neck rotation. Figures 13 and 14 depict the reduction in strike envelope determined experimentally during the 1979 Army tests. Figure 13 is for the conventional MIL-S-58095 restraint and Figure 14 shows the improvements when using the IBAHRS. Both tests were conducted at the 95th percentile potentially survivable crash pulse. Results on this complete test series are reported in Reference 8.

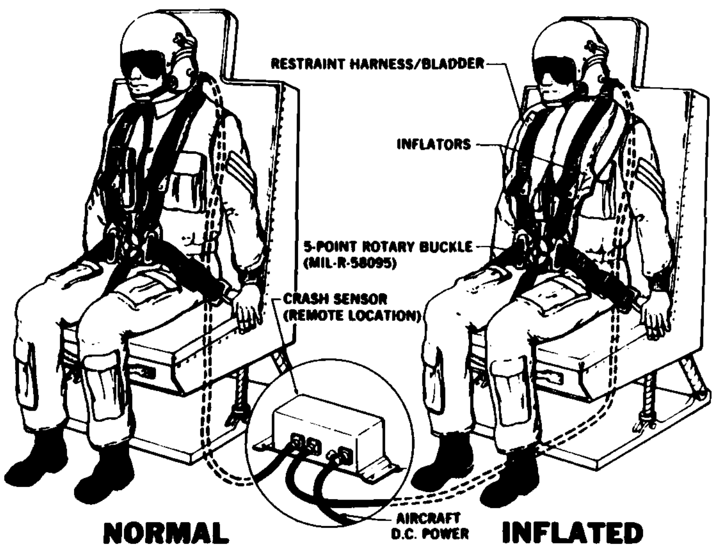

Figure 12. Inflatable Body and Head Restraint System (IBAHRS).

The activating crash sensor for the IBAHRS must be tailored for the particular aircraft application, and it must not allow triggering during routine flight maneuvers or from vibratory or gust loads or hard (autorotative) landings. The crash sensor threshold accelerations for the Army's AH-64 attack aircraft are shown in Figure 15 (from Reference 9) for G loadings directed along the vertical and longitudinal axes. It is obvious, however, that each aircraft type and mission scenario would have to be examined carefully before selecting the crash sensor characteristics for an IBAHRS. For example, an aircraft with 6 G sustained turn capability could not use the AH-64 crash sensor.

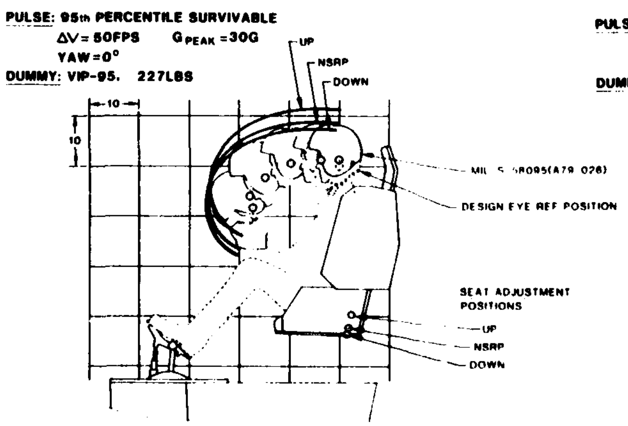

Figure 13. AH-64 Copilot/Gunner Strike Envelope With MIL-S-58095 Restraint.

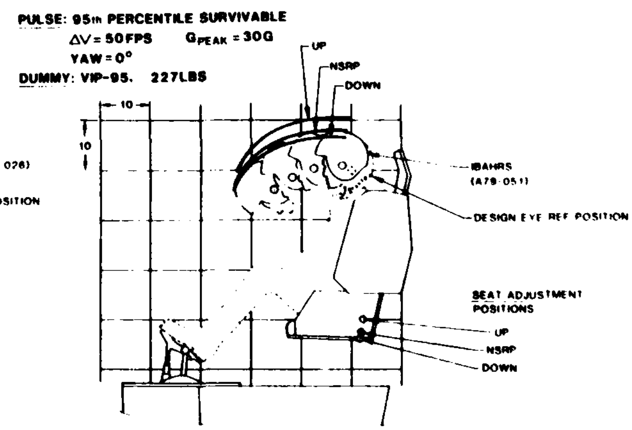

Figure 14. AH-64 Copilot/Gunner Strike Envelope With IBAHRS.

The IBAHRS is currently proceeding through detailed engineering development tests. It will become the standard restraint system for the AH-64 and AH-1S attack helicopters after production approval is given.

Crashworthy Cyclic Control Stick

The floor-mounted, rigid cyclic stick has been a cockpit strike hazard for many years. It has been a major contributor to scores of head injuries and fatalities. A survey of

Army accidents involving 4550 occupants indicates that over 36 percent of the 456 fatalities were due to head and face injuries (Reference 2, Vol. II).

In 1982 the US Army initiated a research effort to develop a cyclic control stick capable of meeting the military specification requirements for normal and emergency control loads as well as having breakaway/telescoping characteristics when struck from above by the crewman in a hard landing/crash. The design is to be generic for retrofit in both the AH-1 and UH-60 aircraft. The vertical separation/collapsing load will be 100-150 pounds, which is within human tolerance for head and face contact, based on a 1 square inch contact area. A grip with 4-inch height adjustment range is also to be incorporated, which is a feature that is not present on existing models.

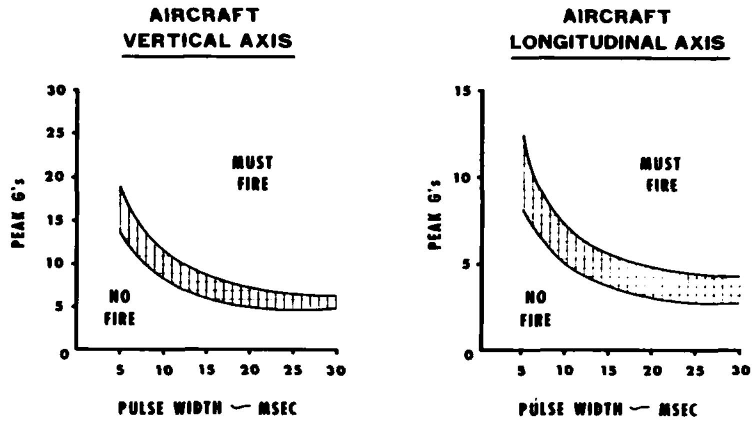

Figure 15. AH-64 IBAHRS Crash Sensor Thresholds.

Whatever the nature of the prototype cyclic stick design, it must first be structurally capable of controlling the aircraft under all conditions. Static pull testing will be performed on the prototype model to assure that a minimum of 200 pounds fore/aft load and 100 pounds lateral load can be withstood. Dynamic tests will then be performed to determine vertical breakaway loads and corresponding reactive head accelerations for the current production sticks and the prototype model. These will consist of simple pendulum tests with the mass and texture of the human head to be simulated at the striking end. Stick impact strike velocities of 30 and 20 ft/sec have been predicted by program SOM-LA for 95th and 50th percentile male occupants, respectively. The pendulum testing will be conducted using these impact velocities.

In future aviation weapon systems development, it is likely that advanced control systems will incorporate sidearm controllers and thus effectively eliminate the conventional cyclic stick as a cockpit strike hazard. In the meantime, retrofit of a well-designed crashworthy stick may be a very cost-effective approach to eliminating a major cause of crewman injuries and fatalities.

Advanced Crashworthy Landing Gear

Load-limiting landing gear are essential to accomplishing crashworthiness goals. From purely an economic viewpoint, the payoff from design for crashworthiness is primarily from reduction of aircraft mishaps, thus enhancing mission effectiveness through greater aircraft availability and avoidance of mishap costs. The MIL-STD-1290 requirement that the landing gear prevent fuselage/ground contact for impact velocities of at least 20 ft/sec for ±10 degrees roll and +15 degrees to -5 degrees pitch attitude, and combinations thereof, is the most significant factor in the realization of a cost-effective return on investment of design for crashworthiness. A landing gear that will prevent fuselage/ground contact at higher impacts than 20 ft/sec is certainly desirable from an aircraft damage prevention standpoint; however, this capability must take into account potential adverse system effects such as:

- Excessive landing gear and attachment weight with attendant decrease in aircraft performance

- Insufficient injury reducing energy attenuation in aircraft structure and seats for the case of impact with a retractable crashworthy gear in the retracted position

- A design that will result in excessive damage to dynamic components during landing gear load attenuation (This may be especially critical for the tilt prop/rotor concept.)

The required roll and pitch attitudes for impact without fuselage/ground contact evolved from a review of the survivable and partially survivable US Army rotary-wing accidents from January 1972 to December 1982. Figure 16 shows the roll frequency of occurrence when only impacts of ±25 degrees roll or less are considered (74 percent of all accidents). Figure 17 shows the pitch frequency of occurrence when only impacts of ± 30 degrees pitch or less are considered (92 percent of all accidents).

Although the AH-64 APACHE has entered limited production, the only fully operational helicopter with landing gear designed to the criteria of MIL-STD-1290 is the UH-60A BLACK HAWK. This landing gear configuration shown in Figure 18 consists of a main and tail gear, each having two stages and a trailing arm for increased stability during impacts with a longitudinal velocity component. The first stage comes into play for normal landings and hard landings up to 20 ft/sec. The second stage contributes to the total system energy absorption during crash impacts at vertical velocities up to 42 ft/sec.

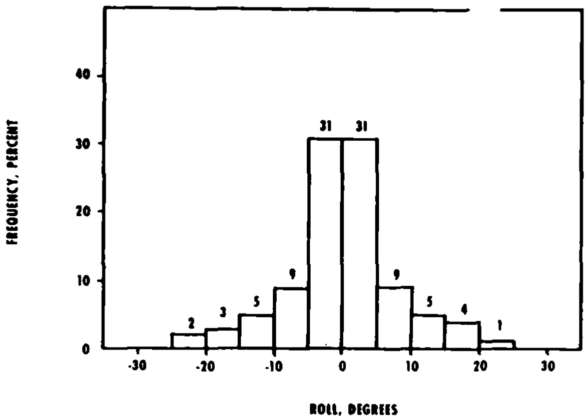

Figure 16. Impact Roll Angle Frequency.

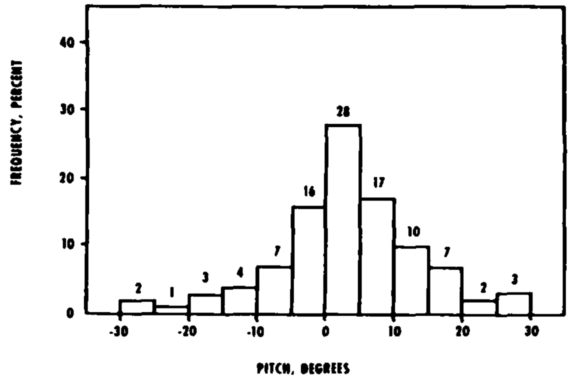

Figure 17. Impact Pitch Angle Frequency .

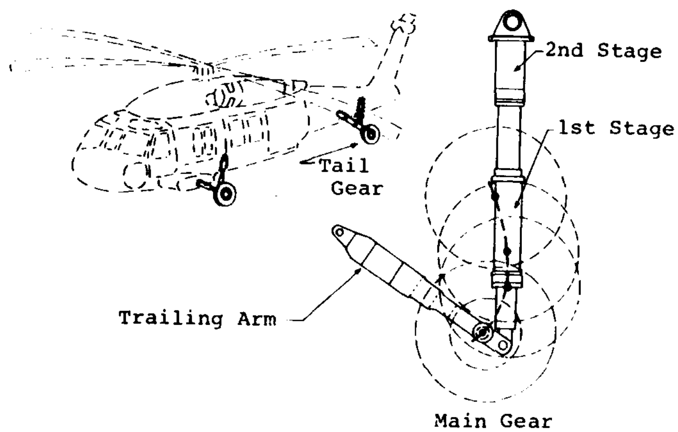

Figure 18. UH-60A Landing Gear .

Additional research is underway to critically analyze crashworthiness design criteria in light of cost and aircraft performance effects when one considers new-generation aircraft that may have retractable landing gear consisting of composite as well as metal components. The emergence of new high performance aircraft such as the tilt prop/rotor which has a higher disc loading than conventional helicopters may dictate the need for new landing gear design criteria.

Using accident rates and mean loss data furnished by the US Army Safety Center, an assessment was made of the benefits of incorporating into the AH-64 a landing gear capable of preventing fuselage-ground impact for vertical sink rates up to 20 ft/sec as compared with using a skid gear comparable to those of the AH-1, UH-1, and OH-58 helicopters. For an AH-64 fleet size of 500, it was calculated that there would be a 14-percent reduction in the accident rate, representing an estimated savings of nearly 570 million dollars for a 20-year fleet life cycle.

Crashworthy Fuel System

In the 1960's, postcrash fires were responsible for nearly 40 percent of all Army rotary-wing fatalities in potentially survivable accidents. In an effort to find a solution to this tragic problem, the US Army Applied Technology Laboratory conducted extensive efforts aimed at developing a crashworthy fuel system (CWFS) for Army helicopters. Particular attention was given to the derivation of fuel tank (bladder) material that was cut, tear, and rupture resistant while incorporating ballistic tolerance characteristics. A CWFS was developed that consisted of self-sealing breakaway valves/couplings; frangible attachments; self-sealing fuel lines; vent valves; cut, tear, and rupture resistant bladders; and a means of preventing postcrash fuel spillage at all postcrash attitudes. Due to the seriousness of the problem, the Army approved fleet retrofit of all helicopters with a CWFS as a safety issue.

With the advent of the Army crashworthy fuel system, postcrash fire statistics have been altered dramatically. During the past 12 years the incidence of thermal fatalities and injuries for CWFS-equipped helicopters has been essentially nonexistent. For example, during the period April 1970 to June 1976, for helicopters not equipped with a CWFS there were 65 thermal fatalities compared to just 1 for helicopters equipped with a CWFS. This is based on nearly the same number of accidents for each case. Since 1976 there have been no fatalities attributed to thermal injuries in potentially survivable accidents of Army helicopters. The highly successful application of this crashworthiness design feature not only has resulted in the prevention of numerous fatalities and a large loss of materiel but has had a very positive effect on aviator morale. In the development of a specification for a new aircraft system, some MIL-STD-1290 design criteria are scrutinized for applicability. However, this is not the case for criteria dedicated to crashworthy fuel systems.

YAH-63 Full-Scale Crash Test

Since the early 1960's, the Army has conducted a series of 41 full-scale crash tests of fixed- and rotary-wing aircraft. The objective of these tests has been to measure,

under controlled conditions, the dynamic structural and occupant response to a variety of crash parameters.

During July 1981 the Applied Technology Laboratory, in conjunction with the NASA-Langley Research Center (LRC) and the US Navy, conducted a full-scale crash test (T-41) of a YAH-63 attack helicopter. This prototype twin-engine, 15,000-pound gross weight class aircraft was acquired by ATL as residual hardware following the Advanced Attack Helicopter (AAH) fly-off competition. The crashworthy design of the YAH-63 is considered representative of that found in the Army's production advanced attack helicopter, the AH-64 APACHE. It is significant that this was the first crash test of an Army aircraft designed from its inception to incorporate most of the MIL-STD-1290 crashworthiness requirements, and it presented a unique opportunity to assess the effectiveness of actual crashworthiness design applications. Specific crashworthy features/experiments on T-41 were:

- Two-stage air/oil crashworthy landing gear
- Controlled crush belly structure
- Production AH-64 load-limiting crewseat incorporating 12-inch maximum stroke
- Developmental joint Army/Navy Inflatable Body and Head Restraint System (IBAHRS) on front crewman
- Prototype Integrated Helmet and Display Sight System (IHADSS) on front crewman
- Crashworthy fuel system including tanks, lines, fittings
- Tie-down strength of high-mass items sufficient for survivable crash loads
- Developmental Navy Flight Incident Recorder/Crash Position Locator (FIR/CPL)
- Developmental Army Accident Information Retrieval System (AIRS)
- NASA experimental package of Emergency Locator Transmitters (ELT's) and crash sensors

The crash test was accomplished using the cable swing/drop method at the NASA-LRC Impact Dynamics Research Facility. Cables were rigged to simulate a 50-ft/sec resultant impact vector of 95th percentile severity. The planned impact conditions were 30-ft/sec longitudinal velocity, 40-ft/sec vertical velocity, and 10 degree nose-up pitch attitude. Figure 19 shows the aircraft in its pretest pull-back position. Due to overestimates of aircraft drag, actual impact conditions were considerably more severe. The actual impact vectors were 36.2-ft/sec longitudinal velocity, 48.0-ft/sec vertical velocity for a 60.1-ft/sec resultant vector. As a result, the impact contained 44 percent more energy than the planned 95th percentile value which places it in the nonsurvivable range. A still sequence photo taken approximately 150 milliseconds after tail contact is shown as Figure 20.

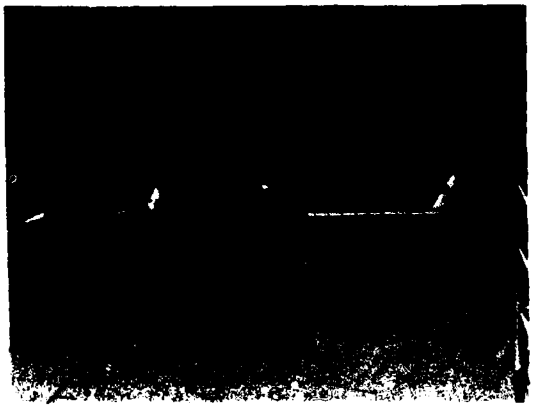

Figure 19. YAH-63 in Pull-Back Position Prior to Crash Test.

Figure 20. YAH-63 Approximately 150 Milliseconds After Impact.

The production bulkhead-mounted AH-64 crewseat traveled through its entire 12 inches of available stroke and "bottomed" on its stops due to the excessive vertical energy. The accelerometer traces which recorded the seat mounting bulkhead and the seat pan vertical accelerations are overlaid in Figure 21. Note particularly the seat pan bottoming pulse which remained above the 23 G Eiband criteria for 17 milliseconds. A dynamic response index (DRI) of 23 was later calculated. Ejection seat relationships established by the US Air Force indicate that this corresponds to a greater than 50 percent probability of spinal injury for the forward located copilot/gunner occupant. Detailed data relating to this test, along with a correlation of predictive versus actual results from use of computer program KRASH, are available in Reference 10.

In summary, the crashworthy landing gear, crushable structure, stroking seats, crashworthy fuel system, and high mass component retention all functioned successfully; and had the desired impact velocities (95th percentile survivable velocities) been obtained, non-injurious accelerative loadings would have been realized by the occupants.

UH-60A BLACK HAWK CRASHWORTHINESS EXPERIENCE

The UH-60A BLACK HAWK helicopter is the first operational Army helicopter designed from inception for crashworthiness using the requirements of the "Crash Survival Design Guide." The results have been dramatic, as evidenced by the following summary of a recent Class A UH-60A mishap:

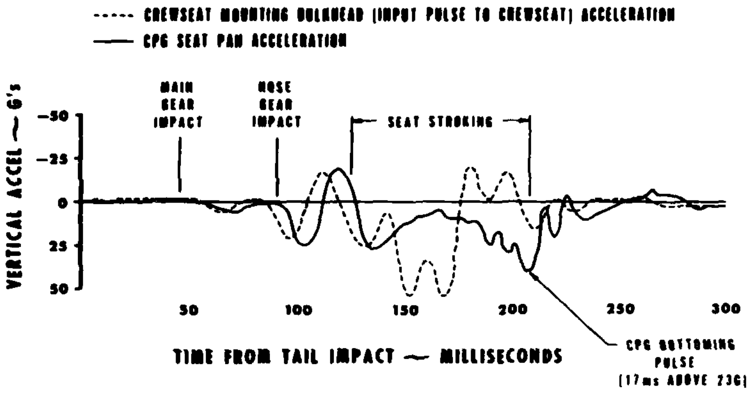

Figure 21. Copilot/Gunner (CPG) Bulkhead and Seat Pan Vertical Accelerations.

The aircraft crashed approximately 20° nose high with a horizontal velocity of 34 ft/sec and a vertical velocity of 49 ft/sec, giving a resultant velocity of 60.4 ft/sec, which for a non-crashworthy aircraft is considered a nonsurvivable impact. The impact sequence is shown in Figure 22.) The aircraft then rebounded with left yaw and right roll until resting on its right side up against a tree. The performance of the energy-absorbing tail and main landing gear and the stroking energy attenuating crew seats, coupled with the structural design for high mass component retention, resulted in maintaining a protective shell around the pilot and copilot and keeping the acceleration loadings in the cockpit below injurious levels. After the crash sequence, the copilot walked away from the aircraft with minor abrasions. The pilot suffered a broken leg and elbow as a result of flailing contact with the cyclic stick and seat wing armor, respectively. The crashworthy fuel system performed perfectly, which in this case was lifesaving in that the right side facing gunner seat occupied by the crew chief failed, resulting in critical injuries to this individual. A design modification is underway to improve the stroking gunner seat.

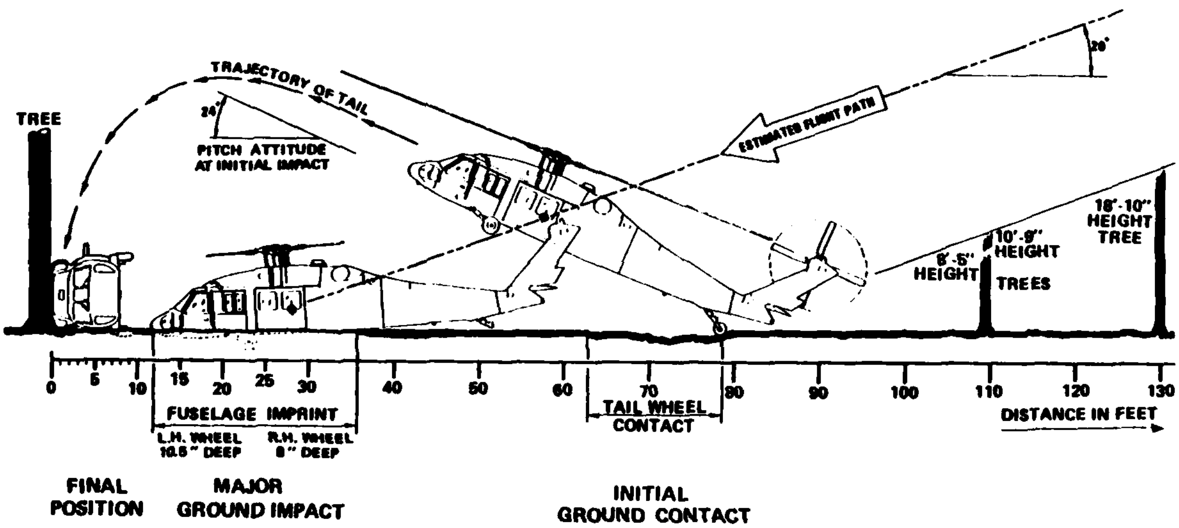

Figure 22. Crash Impact Sequence.

The UH-60A crashworthiness demonstrated in this mishap is remarkable and substantiates that design for crashworthiness will prevent many fatalities and injuries, and the loss of materiel during the life cycle of the BLACK HAWK fleet.

WEIGHT IMPACT AND EFFECTS ON LIFE-CYCLE COST

The many benefits realized by enhancing aircraft crashworthiness are not obtained without some impact on the weight of the aircraft. As discussed previously, this impact can be minimized during the development of completely new aircraft designs as compared to retrofitting crashworthiness features on existing aircraft. A brief survey of seven contemporary helicopter designs (a mix of civil and military designs) revealed varying degrees of integral crashworthiness, ranging from partial to nearly complete compliance with MIL-STD-1290. Weight increments attributed solely to crashworthy features fell between 1.59 and 3.68 percent of design gross weight. Within this range, the weight addition due to use of a crashworthy fuel system averaged 1.07 percent of the design gross weight. Thus, all protective features excluding the CWFS averaged 1.64 percent of design gross weight, which is considered to be an extremely small weight increment for such a high potential return in mission effectiveness. Of course, the weight increase due to crashworthiness design is reflected in aircraft system acquisition and operating costs.

From a total aircraft systems design/operational perspective, design for crashworthiness can have a significant impact on life-cycle cost, especially considering the fact that Army helicopters sometimes remain in the operational fleet for 30 years or more. In-house analyses have been conducted to assess the effects of crashworthiness on life-cycle cost, considering such variables as aircraft acquisition cost; cost of incorporating aircraft crashworthiness features; increased operational cost due to weight/performance penalties for the crashworthy features; personnel training cost; cost of crew injuries/fatalities in accidents; and accident-related property damage cost. Depending on the total cumulative flying hours per year for a fleet of helicopters, the break-even point, i.e., the point where the additional costs for incorporating crashworthy design features is offset, can occur in as little as 3 (wartime flying hour rate) to approximately 9 years (current peacetime flying rate). Beyond the break-even point the cost of owning and operating the fleet is reduced as a result of the crashworthy design. Again, these analyses consider only the costs associated with aircraft damage, personnel injuries/fatalities, and property damage. The total "costs" that are associated with increased mission effectiveness as provided by incorporating crashworthy design, although very difficult to define and quantify, have the potential of being highly significant in a positive sense.

RELATIONSHIP TO CIVIL AVIATION

In the civil aviation community, prevention of accidents has always been a high priority. However, even with technological advancements, increased mechanical reliability, improved pilot training, and intensive studies of accident causal factors, accidents do occur. Statistics from Reference 11 indicate that for one decade (1967-1976) the number of general aviation aircraft involved in accidents was equivalent to at least 38 percent of the total US production during that period. Estimates that an aircraft will be involved in an accident over a 20-year life range are as high as 60-70 percent.

Recognizing this accident probability, it makes sense to apply a worthwhile degree of crashworthiness to contemporary design philosophy. Because of differences in mission profiles, civil aircraft are normally flown somewhat differently than Army helicopters. The FAA Technical Center currently has an effort underway to better define the civil helicopter crash environment ("Rotorcraft Crashworthiness Scenarios," FAA Contract DTFA03-81-C-00035 with Simula, Inc., Tempe, Arizona, scheduled for completion in August 1983). The civil helicopter crash environments may not be sufficiently severe to justify using all of the MIL-STD-1290 crashworthiness design techniques that have been addressed in this paper. From a cost viewpoint the easiest to justify might be the use of state-of-the-art restraint and energy absorbing seat systems, although the crashworthy fuel system should perhaps be at the top of the priority listing of needed crashworthy features. As composite airframe structures become more attractive from a cost/weight standpoint, their demonstrated potential (Reference 12) to act as good energy absorbers should not be overlooked. Usually, however, design innovations to benefit crashworthiness will equate to a design in excess of the Federal Air Regulations (FAR's), which are intended as minimum requirements only rather than design goals. FAA Order DA 2100.1 clearly states, "Such standards do not constitute the optimum to which the regulated should strive" (Reference 13).

Finally, not to be overlooked in the civil area is the very real economic savings that can be gained (in concert with crashworthiness) from the inclusion of an energy absorbing (EA) landing gear. The potential Army savings were addressed earlier and would certainly, to a degree, apply in the civil market. Avoided materiel damage from hard landings alone should go a long way toward justifying an EA gear.

Some design practices, such as excellent protective structure around the occupant along with adequate restraint in agricultural aerial application airplanes, are now standard procedure. In time, it is hoped that a variety of meaningful crashworthiness improvements will be providing increasingly higher levels of occupant protection and damage avoidance.

MAJOR PROGRAM NEEDS

Considering the significant potential payoff for designing Army helicopters for improved crash survivability, the difficulty in retrofitting existing aircraft to make them more crash survivable, and the potential for crashworthiness criteria to significantly drive a new aircraft system design, the need exists to:

- Expand the Army aviation crash survivability program to develop more efficient concepts and measures for improving helicopter crashworthiness while having minimum impact on aircraft system weight, performance and cost.
- Continually improve/upgrade crashworthiness design criteria and standards, considering lessons learned from the UTTAS and AAH development programs; lessons learned from BLACK HAWK and APACHE helicopter operational experience; results from the various composite airframe programs; and new VTOL design concepts (e.g., tilt prop/rotor, which is a cross between the pure helicopter and fixed-wing aircraft).

CONCLUSION

- Too many US Army aircrewmen are still being fatally injured in potentially survivable accidents, and the percentage of major injuries and rate of materiel losses are still far too high.
- Technology and design criteria presently exist to significantly reduce these personnel injuries/fatalities and materiel losses associated with helicopter accidents.

- Army aviation mission effectiveness can be significantly enhanced through the application of crashworthiness design to Army helicopters.
- Life-cycle costs can be significantly reduced through the application of crashworthiness design to Army helicopters.
- MIL-STD-1290 has proved to be a viable, cost-effective requirements document.
- Although much higher levels of crashworthiness can be achieved in a complete new helicopter system design, significant improvements can be made in the crashworthiness of existing helicopters through retrofit programs.
- The need exists to continue to develop and apply efficient and economical measures for improving the crash survivability of existing and new-generation helicopters.
- The need exists to continually improve/update helicopter crashworthiness design criteria and standards.
- Military helicopter crashworthy features are directly applicable to the civil/commercial helicopter fleet.

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THE IMPACT OF THE F/A-18 AIRCRAFT DIGITAL FLIGHT CONTROL SYSTEM AND DISPLAYS ON FLIGHT TESTING AND SAFETY

by

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SUMMARY

This paper overviews the development of the digital fly-by-wire Flight Control System (FCS) in the F/A-18 aircraft. A general description of the FCS and an overview of the significant changes that have been incorporated to improve handling qualities and to correct anomalies that were discovered during the Full Scale Development (FSD) program are presented. The interface of the FCS with the total avionics package of the F/A-18 via the 1553 multiplex bus and the impact of this interface on specific flight testing and FCS development is also highlighted. The impact of the flight control laws on high Angle of Attack (AOA) handling qualities, including a discussion of the changes that were made as a result of a spin accident in November 1980, is presented. Throughout the discussion, references to the specialized displays and controls that are implemented in the F/A-18 to assist the pilot and enhance flight testing and safety are discussed.

LIST OF SYMBOLS

α	-	Angle of Attack
ADC	-	Air Data Computer
Ay	-	Lateral Acceleration
$C_{N_{\beta Dyn}}$	-	Directional Divergence Parameter ($= C_{N_\beta} \cos\alpha - \frac{I_z}{I_x} C_{\ell_\beta} \sin\alpha$)
C_{l_β}	-	Lateral Stability Derivative
C_{n_β}	-	Yawing Moment Coefficient
C_m	-	Pitching Moment Coefficient
dB	-	Gain in Decibels of Bandpass Filter
ΔQ_{c_i}	-	Differential Dynamic Pressure
δ_H	-	Stabilator Deflection
δ_R	-	Rudder Deflection
Fs/g	-	Longitudinal Stick Force/g
G	-	N_y Units
I_x	-	Roll Inertia
I_y	-	Pitch Inertia
I_z	-	Yaw Inertia
K	-	Gain of Bandpass Filter
L/D	-	Lift/Drag Ratio
N_y	-	Lateral Acceleration
N_z	-	Normal Acceleration
OAT	-	Outside Air Temperature
P	-	Roll Rate
P_s	-	Static Pressure
q	-	Pitch Rate
	-	Dynamic Pressure
q_c	-	Dynamic Pressure, Indicated
q_{ci}		
R	-	Yaw Rate
S	-	Wing Area
TA	-	Ambient Temperature
TAS	-	True Airspeed
τ, λ	-	Time Constant, sec
τ_1	-	Time Constant in Pade Approximation in Bandpass Filter
τ_2, τ_3	-	Real Roots in Numerator of Bandpass Filter
$\omega_1, \omega_2, \omega_3$	-	Frequency of Numerator and Denominator Terms in Bandpass Filter
ξ_1, ξ_2, ξ_3	-	Damping of Numerator and Denominator Terms in Bandpass Filter

FLIGHT CONTROL SYSTEM DESCRIPTION

The FCS in the F/A-18 employs a full authority, high gain control augmentation mechanization. Early versions of the FCS (3 series and 4 series Programmable Read Only Memory (PROMS)) utilized applied stick forces to generate the electrical inputs required to control the aircraft. A major FCS design philosophy change was implemented in 6.X series and subsequent PROMS so that stick position vice stick force is utilized to generate the electrical inputs which are then routed to the flight control computers (FCC) to be processed through specified control laws to provide desired aircraft response (figure 1).

Primary pitch control is provided by symmetric deflection of horizontal stabilators. Trailing edge and full span leading edge maneuvering flaps provide optimum lift-to-drag ratios for maneuvering, cruise and high AOA flight conditions. In approach configurations, leading edge flaps and rudders (toed-in) are scheduled with AOA to improve longitudinal stability characteristics. Trailing edge flaps are scheduled with dynamic pressure (q_c) to a maximum deflection of 30 or 45 degrees (TED) dependent on flap switch position. The ailerons are symmetrically drooped to match the scheduled trailing edge flap deflection. A speed brake located on the upper surface of the aft fuselage provides drag control in the cruise configurations (Flaps-UP/AUTO). Roll control is provided by conventional ailerons, differential stabilators, and differential deflection of the leading and trailing edge flaps (differential flap deflections are dependent on flight conditions). Directional control is provided by dual rudders. A rolling surface to rudder interconnect (RSRI) is used to improve turn coordination. Also, a rudder pedal to roll command signal is used to improve roll response at

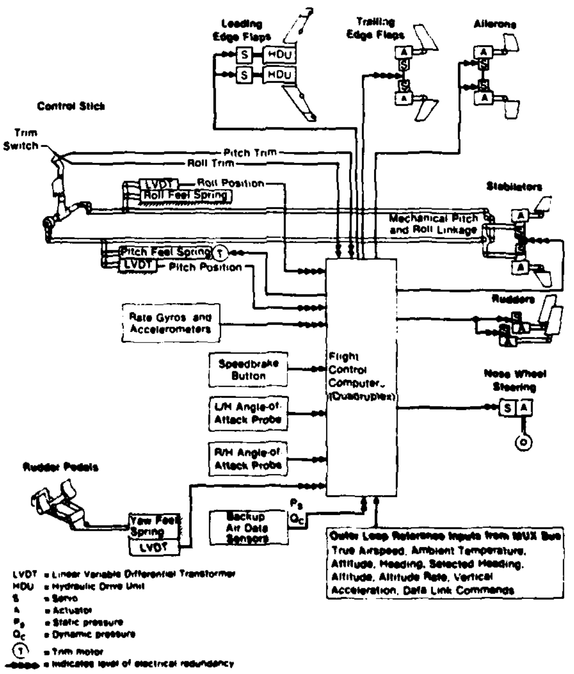


Figure 1 F/A-18 FCS Functional Diagram

higher AOA in cruise configuration and to reduce sideslip excursions in approach configurations. The control laws automatically revert to a SPIN mode if specific yaw rate conditions are exceeded to facilitate recovery from out of control flight. Additionally, pilot relief modes are provided through the autopilot to facilitate weapon system management.

Power to the FCS is supplied by 28 Vdc Power from the aircraft's electrical system. Four independent branches of the hydraulic system provide primary and backup hydraulic pressure to the surface actuators. For three similar failures of motion feedback sensors in a given axis, control is accomplished using a digital Direct Electric Link (DEL) mode, which provides a direct electrical path from the pilot input sensor to the control surface actuator. Should three digital processors fail, longitudinal and roll control is accomplished by a backup mechanical mode to the stabilators. The mechanical controls are conventional cable, push rod, and bellcrank systems. In the mechanical backup mode, stick-to-stabilator gearing is modified by a nonlinear linkage to provide the desired sensitivity between stick forces and deflections for all flight conditions. Aileron or rudder control is available in the mechanical mode through an analog DEL path. In the event of a total electrical failure, only mechanical control of the stabilators is available.

Reliability and maintainability of the FCS have been enhanced during the FSD program by continued improvements to the designed Built-in-Test (BIT), the memory inspect (MI), and the maintenance monitor capability. Additionally, expanded maintenance advisory information is available through the incorporation of BIT Logic Inspect (BLIN) capability. This feature provides the capability to automatically search the flight control computer memory to obtain relevant failure isolation data and display it by channel on the cockpit displays. A more complete description of the BIT, MI, and BLIN features is contained in reference 1.

FLIGHT CONTROL SYSTEM INTERFACE

The FCS interfaces with other avionics in the F/A-18 via a 1553 multiplex bus as shown in figure 2. It is this interface capability which, in conjunction with the unique FCS displays and controls, aided the development and testing of the FCS. The mission computer (MC) FCC interface was designed to allow the FCC's to receive data for outer loop control computations and initiated BIT commands and to transmit sensor data, flight test data, and BIT results to the other avionic components in the aircraft. The BLIN and MI inspect features mentioned earlier are also dependent on the interface capability provided by the 1553 multiplex bus. During the course of FCS development, the MC-FCC interface provided a unique capability for specialized diagnostic testing in the areas of performance, high AOA, and development of the active oscillation controller.

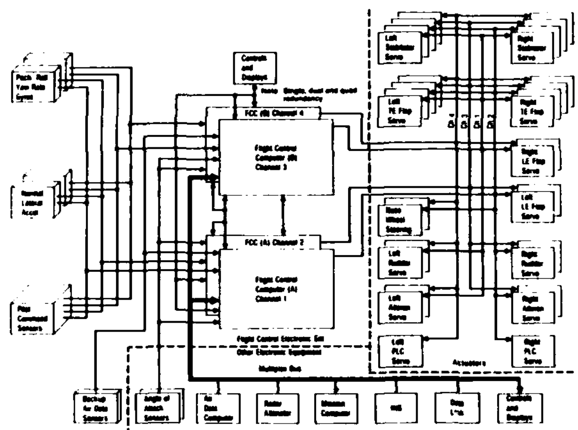


Figure 2 Flight Control Interface

SPECIALIZED CONTROLS AND DISPLAYS

Several specialized displays and controls were incorporated to enhance safety and facilitate FCS testing. FCS controls and displays are shown in figure 3. Special displays that aided in FCS testing included the FCS failure matrix display shown in figure 4 and the specialized SPIN displays discussed later in this paper. The FCS failure matrix display provided the pilot with status information on which FCS shutoff valve or sensor had failed whenever a FCS caution occurred. Additionally, a reset feature was provided via a button on the flight control panel (figure 3). Positive indication of a successful reset for a given failure was provided by removing the X from the FCS failure matrix. This same information was provided via a similar binary display panel in the ground station which displayed FCS status to ground test personnel.

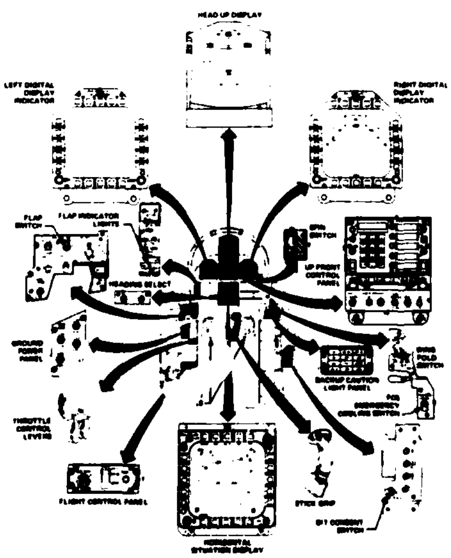


Figure 3 FCS Controls and Displays

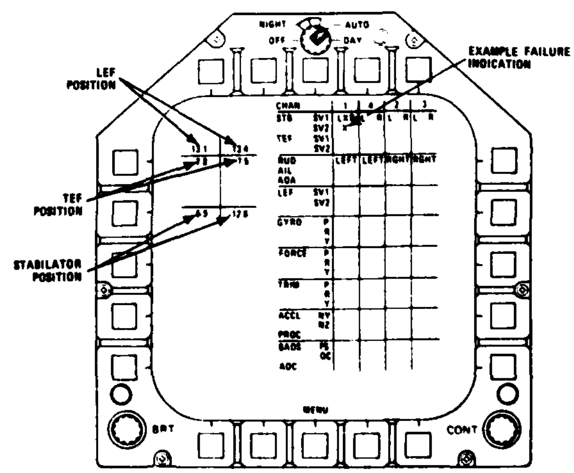


Figure 4 FCS Failure Matrix Display

FLIGHT CONTROL SYSTEM DEVELOPMENT

GENERAL

Throughout the FSD flight test program of the F/A-18 airplane, several updates and changes to the basic control laws that provide the signal shaping between pilot command inputs and resultant control surface commands have been made. The versatility of the digital design of the F/A-18 FCS provides a unique and practical way of implementing any desired control law changes. Control laws are programmed on a number of PROM modules which are mounted on removable boards in each of the two FCC's. Control law changes are introduced by incorporating updated or revised PROM's. Since the first flight in the F/A-18 (18 November 1978), five major PROM series (more than 56 PROM versions) have been evaluated. The major PROM versions tested include 3.X series, 4.X series, 6.X series, 7.X series, and 8.X series PROMS. Table I summarizes the major changes that were incorporated in each of the major PROM series. Control law changes have been incorporated to improve handling qualities at all flight conditions (including high AOA and out-of-control), improve roll performance, reduce structural loads, improve departure resistance characteristics, incorporate and refine pilot relief modes, and provide an active oscillation controller to suppress undesirable in-flight oscillations.

Table I

PROM SERIES VERISON

PROM Version	Reasons for Change	Time Frame
3.X (7 Total) (3.11, 3.12, 3.16, 3.18, 3.19, 3.21, 3.23)	Improve Handling Qualities Improve Carrier Suitability	Nov 1978-Dec 1979
4.X (26 Total) (4.0, 4.1, 4.3.0.X, 4.3.1.X, 4.3.2)	Reduce Time Delays Add RSRI vice SRI Spin Mode Improvements Roll Modifications	Jan 1980-Nov 1981
6.X (4 Total) (6.0, 6.0.1.1, 6.0.1, 6.0.2)	Reduce Time Delays Position vice Force Sensors Autopilot Modes Incorporated	Nov 1981
7.X (14 Total) (7.0, 7.1.X, 7.2, 7.3, 7.4)	Revised Spin Logic Improve Directional Stability AOC Development	Mar 1982
8.X (4 Total) 8.0, 8.1, 8.2, 8.2.1	Throttle Sensitivity Autopilot/APC/ACLS Improvements	July 1982

FCS HIGH AOA CONTROL LAW DEVELOPMENT

Two of the major design goals for the FCS at high AOA were (1) to augment departure/spin resistance and (2) to automatically provide sufficient control authority for recovery from all spin modes. It was also important that FCS air data sensor failures (AOA, Qci, P_s) not degrade high AOA departure/spin resistance or inhibit/prevent recovery from poststall gyrations or spins. During FSD high AOA/spin testing, many changes were made to improve high AOA characteristics and spin recovery capability. On occasion, unexpected FCS response occurred in the high AOA flight region. As a result of these experiences, the FCS has become, at the same time, more complex and more effective in this flight regime. In addition, unique spin recovery cockpit display concepts have been successfully verified that have the potential to significantly increase flight safety. The purpose of this section is to briefly describe some of the design concepts applied in the FCS at high AOA and to relate some of the more significant changes made to the control laws based on test results obtained during FSD flight tests.

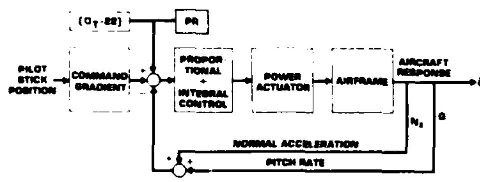


Figure 5 Control Laws - Longitudinal Axis

Longitudinal Axis: A simplified version of current high AOA longitudinal control law mechanization is illustrated in figure 5. Several feedbacks are utilized in the FCS to provide desired high AOA handling qualities characteristics. Normal acceleration feedback provides essentially constant stick force per G at airspeeds above approximately 390 KCAS. Pitch rate feedback is blended with N_z feedback between 390 KCAS and 260 KCAS to improve low airspeed high AOA controllability. Roll rate X yaw rate feedback is utilized to reduce inertia coupling tendencies which were encountered particularly in the 15 to 20 degree AOA region and to reduce vertical tail loads at high airspeed. AOA feedback provides an artificial high AOA stall warning cue by abruptly increasing the stick force per degree g gradient above 22 degrees AOA as illustrated in figure 6. A stall warning tone is generated as AOA increases above 35 degrees. AOA feedback thresholds in order of occurrence have ranged from 23 to 20 to 15 and back to 22 degrees during FSD testing. The design goal was to provide necessary artificial stall warning and at the same time satisfy maneuvering AOA (15 to 30 degrees) handling qualities requirements, particularly in the air combat maneuvering environment (ACM). As an example, the AOA feedback threshold was changed from 15 to 22 degrees due to undesirable Fs/g changes with airspeed. With the threshold set at 15 degrees AOA, it was found that, during simulated ACM evaluations, there was an undesirable change in the stick force per g gradient over a relatively small speed range as illustrated in figure 7. This situation was rectified by shifting the AOA threshold to 22 degrees.

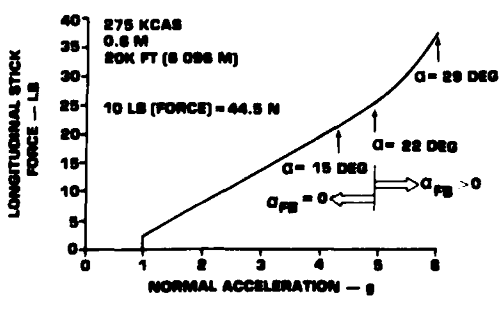


Figure 6 High AOA Feedback Effect on Fs/g

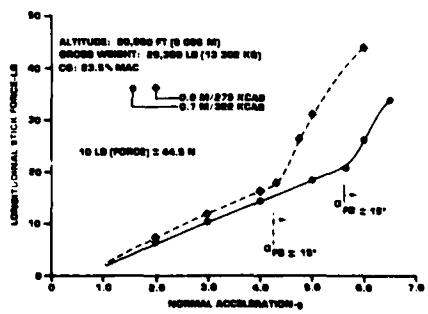


Figure 7 Fs/g Versus Airspeed at 15 Degrees AOA

Maneuvering Flaps: F/A-18 maneuverable LEF and TEF provide increased lift, increased lateral-directional stability, and improved dutch roll damping at high AOA. In the UP/AUTO configuration, LEF and TEF positions are scheduled with AOA and Mach number as illustrated in figure 8. Initial FSD high AOA flight tests focused on lateral-directional stability and control characteristics at AOA's between 15 and 45 degrees. During this testing, maximum LEF deflection was 25 degrees at AOA's above approximately 25 degrees. Results of testing at approximately 35 degrees AOA showed a marked reduction in $C_{N_{\beta_{Dyn}}}$ (Directional Divergence Parameter), as shown in figure 9, due to a significant decrease in lateral stability. The level of $C_{N_{\beta_{Dyn}}}$ at AOA's less than $C_{L_{MAX}}$ was considered unacceptable. As a result, LEF maximum deflection at high AOA was subsequently increased to 34 degrees. As shown in figure 9, the increased LEF deflection provided increased departure resistance. This was obtained as a result of increased lateral stability up to approximately 40 degrees AOA. The trailing edge flap scheduling strongly affects dutch roll damping and L/D ratio. Current F/A-18 control laws command the trailing edge flaps to the full retracted position at maneuvering AOA's approximately 20 degrees.

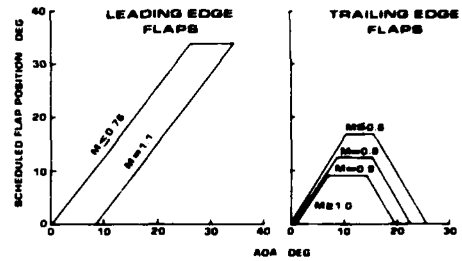


Figure 8 Maneuvering Flap Schedules

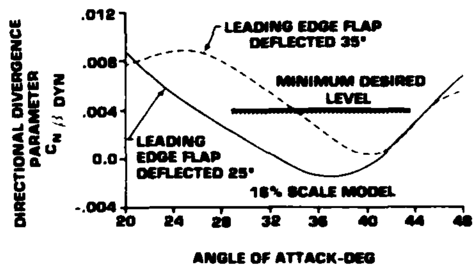


Figure 9 $C_{n_{\beta_{Dyn}}}$ Variation with LEF Position

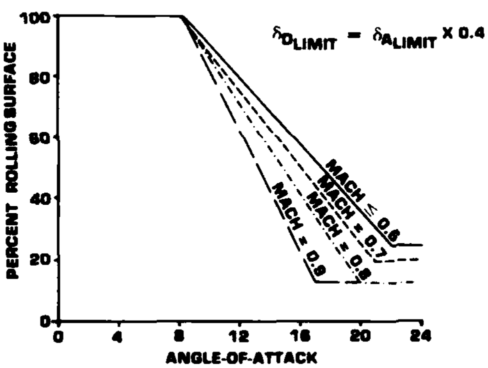


Figure 10 Rolling Surface Authority Versus AOA

Lateral-Directional Axes: Departure and spin resistance is further increased by reducing differential tail and aileron authority at high AOA's (figure 10). The design intent was to significantly reduce the magnitude of adverse yaw with lateral command while still retaining as much coordinated roll response as possible. The F/A-18 FCS incorporates a rolling surface to rudder interconnect (RSRI), functionally similar to an aileron-rudder interconnect (ARI), which provides a proverse yaw contribution during lateral stick inputs, further reducing the adverse yaw tendencies and improving roll coordination (figure 11). A rudder pedal to rolling surface interconnect (figure 12) is also included in the FCS to reduce proverse yaw during rudder rolls. The improved roll coordination minimizes N_z coupling at high AOA due to kinematic coupling (i.e., interchange of AOA and sideslip during uncoordinated rolling maneuvers). Several feedbacks are utilized at high AOA to augment bare airframe lateral-directional stability (see figures 11 and 12): (a) lateral acceleration feedback for increased directional stability, (b) yaw rate feedback for increased directional damping, (c) pα (roll rate x AOA) feedback for improved roll coordination by rolling the airplane about the stability axis (or velocity vector) vice the body axis, (d) roll rate feedback for increased Dutch roll damping, and (e) PQ (roll rate x pitch rate) and PR (roll rate x yaw rate) feedback to reduce inertia coupling tendencies.

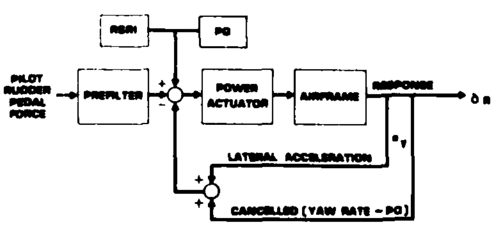


Figure 11 Control Laws - Directional Axis

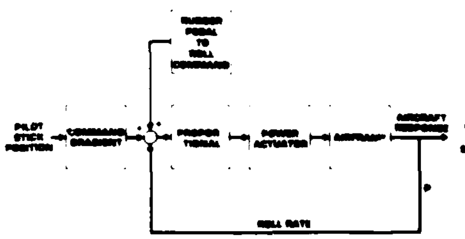


Figure 12 Control Laws - Lateral Axis

<u>FCS Control Law Inertia Coupling Compensation</u>: During FSD high AOA departure resistance testing, negative g departures occurred in the middle of the maneuvering envelope (15 to 20 deg AOA region) when aggravated controls were applied for even brief periods of time (less than 2 seconds). Data analysis indicated that these departures were primarily caused by roll coupling. The combined effect of roll coupling in combination with adverse sideslip led to excessive vertical tail loads and occasionally, to negative g excursions approaching the negative g structural limit. The versatility of the FCS was again demonstrated, in this case, by use of inertia coupling feedback compensation.

An examination of the equations of motion shows that sideslip and g overshoots during rolling pullouts can be reduced if inertial coupling moments can be alleviated. Roll-yaw coupling generates pitch accelerations by roll rate times yaw rate multiplied by an inertia characteristic ratio:

$$q = pr\,(Iz - Ix)/Iy$$

Similarly, roll-pitch coupling causes yaw acceleration by roll rate times pitch rate multiplied by another inertia characteristic ratio:

$$r = pq\,(Ix - Iy)/Iz$$

Therefore, appropriate rudder and elevator deflections were programmed into the flight control PROMS as a function of dynamic pressure and body axis rates to counter yawing and pitching moments due to inertial coupling:

$$\delta r = pq\,(Iy - Ix)/qsb\,C_{n_{\delta r}}$$

$$\delta H = pr\,(Ix - Iz)/qsc\,C_{m_{\delta H}}$$

Figure 13 illustrates the effect of inertia coupling compensation during a full lateral stick input. The magnitude of AOA unloading is reduced considerably when the inertial-coupling compensation is engaged. Full lateral stick rolls were also markedly improved and vertical tail bending moments were reduced to below design values. This proved to be one of several instances where a FCS software programming change solved difficult flying qualities/structural problems that would usually have required a hardware change.

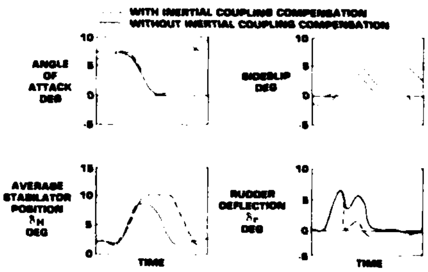


Figure 13 Effect of Inertia Coupling Compensation

<u>FCS Control Law Changes for Improved Departure Resistance</u>: During the later stages of F/A-18 FSD testing, weak directional departure resistance at typical maneuvering AOA was identified. Directional departures occurred at high subsonic Mach number in the 20 to 30 degrees AOA region of the flight envelope, particularly with centerline tank or three external fuel tank loadings. F/A-18 basic airframe plus centerline tank weak directional stability levels at high subsonic Mach number are illustrated in figure 14. Departures at high dynamic pressure flight conditions were of serious concern primarily because of the potential for structural overload of the vertical tails. The result of FCS software changes to correct a significant flying qualities deficiency was again clearly demonstrated. The result of FCS control law changes on departure resistance is illustrated in figure 15. As can be seen, the final control law version (7.1.3 PROMS) was successful in controlling sideslip at high subsonic Mach number which eliminated nose slice departures with symmetric stores loadings.

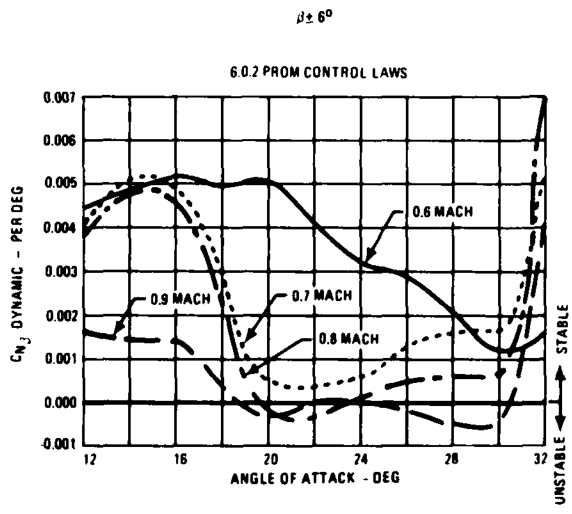


Figure 14 Centerline Tank Loading Mach Number/ AOA Effect on Directional Departure Resistance

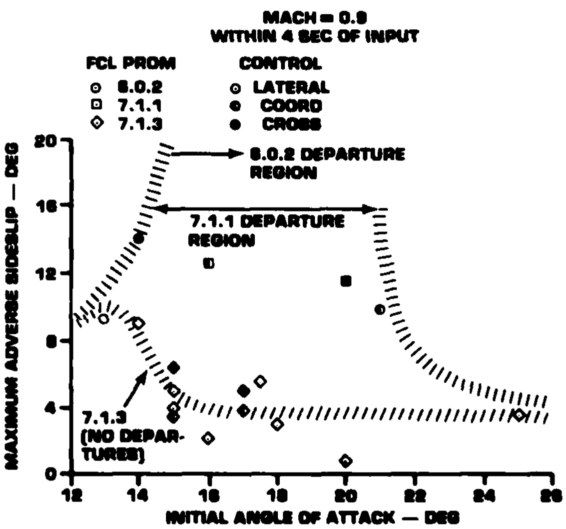


Figure 15 Control Law Effects on Departure Resistance

Evolution of FCS control law changes made to eliminate departures is illustrated in figure 16. The most significant control law changes made were reduced rolling surface authority with increasing AOA and Mach number and significantly increased lateral acceleratioin feedback to rudder gain. The design tradeoff made was reduced roll rate capability at high AOA/high Mach for increased departure resistance.

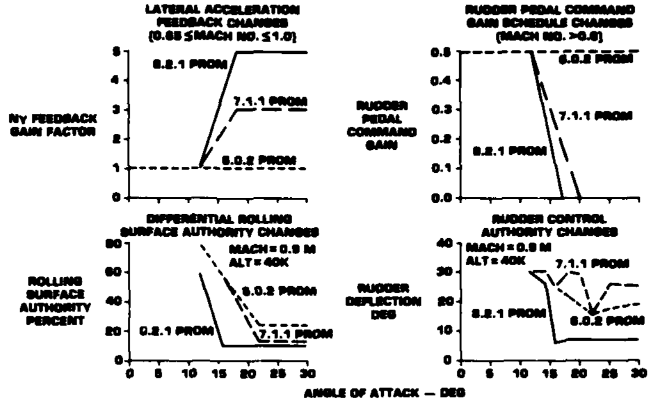


Figure 16 FCS Changes for Improved Departure Resistance at High AOA

FLIGHT CONTROL SYSTEM TESTING

GENERAL

Flight testing of the F/A-18 has involved both the classical stability and control tests (reference 2) listed in table II and specialized frequency sweep techniques (used for equivalent system analysis). Additionally, a major emphasis was placed on the definition and performance of precise mission tasks. A list of the mission tasks evaluated during the development testing of the F/A-18 is presented in table II. The primary evaluation criteria used when evaluating the assigned tasks was the Cooper-Harper Scale reference 3. Handling qualities ratings assigned to these precisely defined tasks have provided the most effective quantifying measurement of the FCS performance during its evaluation. In addition to the handling qualities ratings assigned to mission tasks, the FCS has also been quantified in terms of equivalent system frequencies and damping using a maximum likelihood parameter identification technique reference 4. The use of this parameter identification technique has been very successful in demonstrating the success the contractor has had in reducing the overall system equivalent time delay as the control laws/FCS changes evolved.

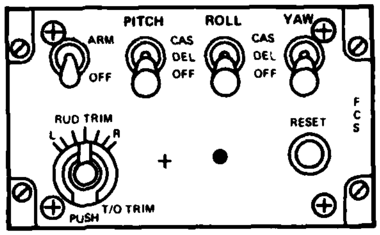
Table II

Classical Flight Test Maneuvers

Maneuver	
Doublet Pitch and Bank Attitude Captures Frequency Sweeps Wind-Up Turn Wind Down Turn Sudden Pull Ups Full Deflection Rolls	
Primary Mission Tasks	
Mission Task	Performance Criteria
Column Formation 300 KCAS/2 g 400 KCAS/4 g	Maintain position (20 feet nose-to-tail, 20 feet stepdown) ±10 feet at 400 KCAS/4g; ±5 feet at 300 KCAS/2 g.
Aerial Refueling 250 KCAS	Engage and maintain a plugged position.
Air-to-Air Tracking 300 KCAS/2 g 350 KCAS/3 g 400 KCAS/4 g	Track a stabilized target at specified flight condition. Maintain pipper ±2 mils on targets tailpipe while maintaining 1500 feet nose-to-tail distance.
Air-to-Ground Tracking 30° Dive 45° Dive 60° Dive	Maintain pipper ±2 mils on target.

Special Flight Control Panel: A special flight test flight control panel was provided in selected FSD aircraft to permit flight testing of the degraded modes of the FCS. The design features of this special flight control panel allowed the pilot to select the DEL or mechanical mode of the FCS in each control axis (pitch, roll, or yaw). Figures 17(a) and (b) show the special flight test flight control panel and defines the mode selected with each switch position. Selection of the different available modes required the pilot to select the desired mode, place the arm switch to arm, then depress the nosewheel steering/designate switch on the control stick. Disengagement from the selected mode and reversion to the normal CAS mode was rapidly available by depressing the autopilot disengagement switch on the control stick.

This same special flight test control panel was also utilized during diagnostic testing of the various roll modifications. Use of the special panel to deselect portions of the roll modifications is also shown in figure 17(c). The significant difference between using the panel for degraded modes and as a diagnostic tool to assess the roll modifications was the position of the arm switch. If the arm switch was on, the appropriate degraded mode was selected. If the arm switch was off, then a portion of the roll modification was inhibited.



Flight Test Modes are engaged by:

1. Select desired pitch, roll, or yaw mode
2. Arm the system.
3. Engage with NWS switch.
4. Disengage with autopilot disengagement switch.

Figure 17(a)

The following conditions can be obtained with the mode switches (6.X series):

Switch Position	Pitch	Switch Roll	Yaw
CAS	Pitch CAS to stabilator.	If yaw CAS is selected: Roll CAS to ailerons. Roll CAS to stabilators if stabilators are not in mechanical. If yaw DEL is selected: Rol digital DEL to ail. Roll digital DEL to stabilators if stabilators are not in mechanical.	Yaw CAS to rudder.
DEL	Pitch digital DEL to stabilators.	Digital DEL to ailerons. Roll DEL to stabilators if stabilators are not in mechanical.	Yaw digital DEL to rudders.
OFF	Stabilators to mechanical.	Roll CAS to stabilators if stabilators are not in mechanical. Analog DEL to ailerons and rudders.	Not available switch position blocked.

Figure 17(b)

Ft Panel Switch Position	Arm Switch Position	Feature Selected[1]
PCAS - OFF	OFF	Reduces total differential stabilator to ± 20 deg.
PDEL	OFF	Disengage differential LEF.
RCAS - OFF	OFF	No reversed aileron gain or aileron disengaged.
YCAS - OFF	OFF	Differential TEF disengaged.

NOTE: (1) Features can be selected individually or all combination.

Figure 17(c)

"Fixed Flap" Mode: The "fixed flap" mode was originally intended to provide a means to optimize the maneuvering flap schedules for cruise performance. The cabability was also used to develop flap schedules to improve approach airspeeds and to improve departure resistance and spin recovery during the high AOA test program. A modified version of the "fixed flap" mode was used to develop the active oscillation controller. The basic design and operation of the "fixed flap" mode demonstrates the flexibility and versatility of the FCC-MC interface in the F/A-18. The MC was programmed to accept a 4 X 2 matrix of data information. In the case of the performance and high AOA testing, this 4 X 2 matrix consisted of four pairs of leading and trailing edge flap commands. In the case of the active controller development, these fixed pairs of data corresponded to gain and phase shifts to a nominal bandpass filter. Operation of the fixed flap mode is summarized in figure 18(a). Changes to the programmed settings could also be made by the pilot or ground personnel via the Up Front Control as described in figure 18(b).

OPERATION OF "FIXED FLAP MODE"

1. Arm the "fixed flap" mode via UFC.

2. Select A, B, C, or D setting on FCES display (hold button until "ARM" cue is displayed).

3. Select fixed flap via nosewheel steering/designate button (look for 1234 on display).

4. Deselect via autopilot disengage switch.

NOTE: Four data pairs (A, B, C, and D) can be changed by pilot via UFC in flight or on ground.

Figure 18(a)

CHANGING FIXED FLAP DATA IN MISSION COMPUTER

1. Arm Fixed Flap Mode
   Menu BIT MI Unit 28
   Address 31214
   Data Option
   Address 27650 Enter 1

2. Address Appropriate 6 Digit Code for A, B, C, or D setting.

3. Enter 6 Digit Octal Code for Desired Flap Setting.

Figure 18(b)

These changes could be implemented on the ground or in flight. The mode could be quickly disengaged by depressing the autopilot disengage switch (paddle switch) on the control stick. Safety was also enhanced by requiring the fixed flap mode to be activated by a discrete input and by providing positive feedback to the pilot via the special FCS display (figure 18(c)).

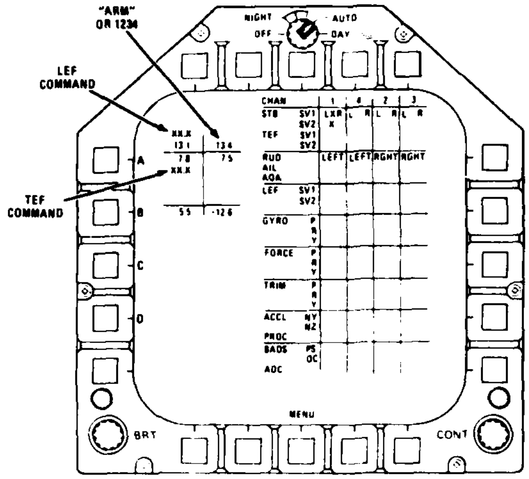


Figure 18(c)

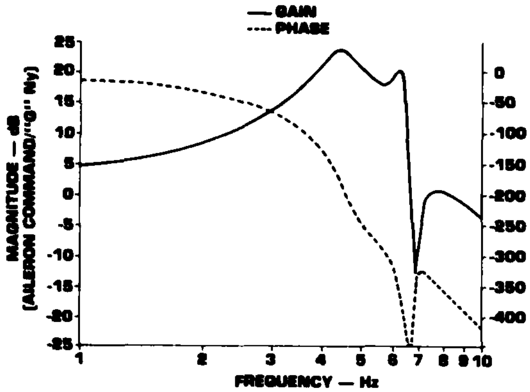
TESTING THE ROLL MODIFICATIONS

One of the two areas of interest during the flight test program that particularly demonstrated the utility of the special flight test control panel was the testing of the roll modifications. As a result of roll performance testing conducted in the F/A-18 with 3 series and 4.1.X series PROMS, several modifications and control law changes were incorporated to improve roll performance in the low altitude transonic flight regime. Initial roll modifications included increasing the wing stiffness, extending the ailerons to the wingtip, incorporating differential trailing edge flap capability ($\pm$8 degrees), increasing differential stabilator deflection from 20 to 26 degrees, and reversing aileron deflection at high dynamic pressure flight conditions. Use of the special flight test panel described previously allowed diagnostic testing to determine the effect on roll performance of the differential trailing edge flaps, the increased differential stabilator authority, and the reversal of aileron commands at high dynamic pressure flight conditions. Results of this initial diagnostic testing provided the data that led to the decision to delete the aileron reversal and decrease the differential stabilator authority in subsequent PROM versions. Subsequent to the initial roll modification testing, further improvements in roll performance in the high transonic, low altitude flight regime were still required. Further testing involved investigating the use of differential leading edge flaps to enhance roll performance in the area of interest. The decision to pursue this course of action was prompted by analysis which indicated that the reduced roll performance at high transonic, low altitude flight conditions was attributable to wing twisting at these high q_c flight conditions. Differential deflection of the leading edge flaps reduced the adverse wing twist and resulted in improved roll performance. Initial testing involved performing 360 degree rolls with the outboard leading edge flaps prerigged to a 6 degree differential deflection. These test results proved favorable and a prototype system was designed and implemented in an FSD aircraft. Testing was accomplished on the prototype system with 4.3.2 PROMS. Diagnostic testing of the effect of the differential leading edge flaps was also possible through the special flight test control panel (figure 17(c)). Results of the diagnostic testing of the various roll modification evolved into the final production roll improvement package which consisted of increased wing stiffness, extended ailerons, and differential trailing and leading edge flaps. Provisions for the production roll improvements were incorporated in the 6.X and subsequent PROM versions.

DEVELOPING THE ACTIVE OSCILLATION CONTROLLER

The area of flight testing that particularly demonstrated the unique flexibility provided by the "fixed flap" capability was the development of the active oscillation controller. During the flutter test program with external stores, objectionable low amplitude directional oscillations (5.6 Hz) were observed at high speed/low altitude flight conditions when heavy stores (MK 80 series bombs) were carried on the outboard weapon stations (stations 2 and 8) with AIM-9's on the wingtip stations. This phenomena was attributed to an asymmetric store pitch mode that coupled with a lateral fuselage bending mode to produce the resultant airplane directional response and lateral acceleration oscillations perceived by the pilot. Analysis of the phenomena indicated that the oscillations were affected by the presence of AIM-9's on the wing tip weapon stations (stations 1 and 9), leading edge and trailing edge flap deflections, and aileron deflections. Based on these observations, early fixes concentrated on scheduling the leading edge flaps and ailerons to effectively eliminate or reduce the magnitude of the oscillations. This approach resulted in schedules that positioned the leading edge flaps 3 degrees leading edge up and the ailerons 4 degrees trailing edge up. These schedules were implemented in the 6.0.2 PROMS and successfully reduced, but did not eliminate, the occurrences of the 5.6 Hz oscillations. The resultant LEF flap schedule also created problems with in-flight loads and leading edge flap operation. Subsequently, a decision to explore an active oscillation control (AOC) mechanization was made. The AOC concept involved using signals from existing FCS sensors to drive the control surfaces to damp out the objectionable 5.6 Hz oscillations. Initial flight test developement involved utilizing a modified flutter exciter control unit (FECU). This FECU was used during the flutter test program to develop an analog filter to suppress the undesired oscillations. Provisions for pilot selectable phase and gain, selectable forward or aft sensor package input, selectable sensor input (lateral accelerometer, yaw rate gyro, or roll rate gyro), and selectable control surface (rudder or aileron) were implemented to suppress the oscillations. Results of this testing confirmed the feasibility of using an active controller to suppress the oscillations. Subsequently, an analog system was developed that used the forward sensor package lateral accelerometer signal to drive the ailerons at an appropriate gain and phase to effectively suppress the 5.6 Hz oscillations.

The basic analog system developed was then implemented in a digital form into a set of flight control law PROMS (7.1.3.1). The general form of the filter implemented into the production PROMS including a corresponding bode plot is presented in figure 19. A pilot selectable Dial-a-Gain and Dial-a-Phase capability was also implemented via the "fixed flap" mode discussed earlier. The Dial-a-Gain and Dial-a-Phase capability provided a means to fine tune the gain and phase of the filter implemented in the control laws. The first change resulting from testing the initial digital design was to change from a 7th order to a 5th order bandpass filter to provide less phase variation across the frequency range of interest (5 to 6 Hz). Additional refinements were required to optimize the filter for a range of external loadings. A decision to activate the system only for MK 80 series bombs on the outboard stations was made since test results with lighter external stores showed that the active controller tended to amplify instead of attenuate the oscillations. This required interfacing the controller with the stores management set via the MC and 1553 multiplex bus. The final filter implemented represents a compromise for the MK 80 series bombs external loadings tested that still provided adequate suppression of the 5.6 Hz oscillations.



$$\frac{K\left(\frac{s}{\tau_1}+1\right)\left(\left(\frac{s}{\omega_1}\right)^2+\left(\frac{2\xi_1}{\omega_1}\right)s+1\right)}{\left(\frac{s}{\tau_1}+1\right)\left(\left(\frac{s}{\omega_2}\right)^2+\left(\frac{2\xi_2}{\omega_2}\right)s+1\right)}\left(\frac{\left(\frac{s}{T_2}+1\right)\left(\frac{s}{T_3}+1\right)}{\left(\frac{s}{\omega_3}\right)^2+\left(\frac{2\xi_3}{\omega_3}\right)s+1}\right)$$

NOTE: FINE TUNING THE FILTER INVOLVED CHANGING THE K AND τ_1 VALUES VIA THE "DIAL-A-GAIN" CAPABILITY

Figure 19 General Form of the Production Bandpass Filter

FCS SPIN RECOVERY MODE DESIGN EVOLUTION

INITIAL CONCEPTS

During the initial design stages of the F/A-18, a great deal of emphasis was placed upon achieving a design that would possess a high degree of departure and spin resistance. The YF-17 which was the prototype for the F/A-18 was known to possess excellent high AOA flying qualities and, as such, the basic aerodynamic design was chosen as a basis for that of the F/A-18. A comparison of the original and current F/A-18 aerodynamic configuration is illustrated in figure 20. The normal operating mode of the FCS is the Control Augmentation System Mode (CAS). The CAS Mode augments the natural aerodynamic stability via control surface authority limiting and use of feedback control concepts.

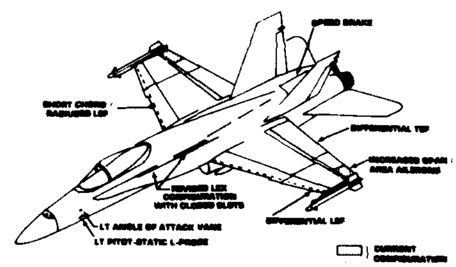


Figure 20 F/A-18 Aerodynamic Configuration

Although lateral-directional control law features in the CAS mode enhance departure/spin resistance, they also reduce control power available for spin recovery. For this reason, an automatic spin recovery mode (ASRM) was incorporated in the FCS. When engaged, the ASRM provides the pilot with full control surface authority regardless of the AOA and opens all feedback loops to provide full antispin control authority for spin recovery.

<u>Automatic Spin Recovery Mode - Initial Design</u>: Establishing safe automatic spin recovery mode (ASRM) engagement/disengagement thresholds was a major consideration before commencement of actual flight tests. The design goal with respect to ASRM logic was to establish engagement thresholds which were not so low as to reduce departure/spin resistance but not so high as to prevent recovery from a spin. Disengagement logic was designed such that the FCS would revert to CAS (i.e., the normal operation FCS mode) during the final stages of spin recovery. While in the ASRM, all feedbacks and control surface limits are removed to provide maximum antispin control authority. It is important to note that pilot spin recovery control inputs are still required in the ASRM. The ASRM does not automatically apply antispin control inputs. During the F/A-18 high AOA/spin FSD program, ASRM logic required changes as more knowledge was gained on F/A-18 spin modes. Initial ASRM engagement/disengagement logic was as shown in figure 21.

- **ENGAGEMENT — YAW RATE ≥ 35 DEG/SEC; TIME ≥ 5 SEC**
- **DISENGAGEMENT — YAW RATE ≤ 15 DEG/SEC**

Figure 21 Original ASRM Logic

During the early stages of the high AOA program, the primary focus of testing was evaluation and/or verification of strong departure and spin resistance. Concurrent National Aeronautics and Space Administration (NASA) F/A-18 drop model spin test results established a requirement to increase the ASRM engagement/disengagement yaw rate thresholds. The basis for this change was a MIL-F-8785B specification requirement that departure resistance be determined by holding sustained prospin control inputs for at least 15 seconds. NASA model testing showed that with ASRM 35/15 degree/second yaw rate engage/disengage thresholds a potential for inadvertent ASRM engagement existed when sustained prospin controls were held for 15 seconds. As a result, ASRM engage/disengage thresholds were increased as shown in figure 22.

- **ENGAGEMENT — YAW RATE ≥ 50 DEG/SEC; TIME ≥ 5 SEC**
- **DISENGAGEMENT — YAW RATE ≤ 30 DEG/SEC**

Figure 22 Revised ASRM Logic

F/A-18 SPIN ACCIDENT

On 14 November 1980, an F/A-18 crashed as a result of a departure that progressed into a low yaw rate spin, which apparently had yaw rates with magnitudes which were less than required to engage the ASRM (50 degrees/second yaw rate). The FCS remained in the Control Augmentation System (CAS) mode. Consequently, insufficient control authority was available for the pilot to achieve recovery. To prevent reoccurrence of these conditions, a cockpit mounted spin recovery mode switch was installed which permitted manual engagement of the spin recovery mode. The manual spin recovery mode (MSRM) switch was installed as an interim fix until such time that the safety and effectiveness of new automatic spin recovery mode logic could be verified. The MSRM switch was also installed in the spin test airplane to permit intentional spin testing for determination of optimum spin recovery control procedures.

Spin Accident Ramifications: The loss of an F/A-18 in an apparent low yaw rate spin had a dramatic impact on the subsequent course of the high AOA test program. Prior to the accident, the major emphasis of testing, as previously noted, was on evaluation of departure and spin resistance. Postaccident testing was expanded to identify all spin modes and to determine optimum spin recovery techniques. In retrospect, it is clear that because of the F/A-18 spin accident and subsequent FSD spin testing, significantly more is known about F/A-18 spin modes, spin recovery characteristics, and operation of the FCS at high AOA than would be otherwise. In particular, as a result of intentional spin testing, the low yaw rate spin mode was identified which is believed to have been responsible for the spin accident. In this regard, initial analytical high AOA simulations and spin tunnel model testing did not predict the existence of this mode as illustrated in figure 23. An example of an actual low yaw rate spin is presented in figure 24.

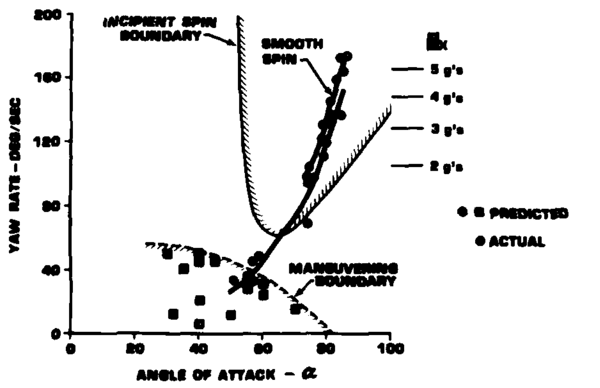


Figure 23 Predicted Maneuver/Spin Boundaries Versus Flight Test

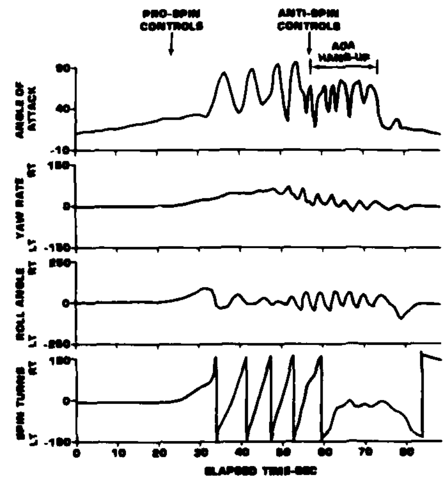


Figure 24 Low Yaw Rate Spin

A contributing factor for no prediction of the low yaw rate spin mode may have been uncertainty in the high AOA aerodynamic data base (rotary derivative data were not available) used to generate the F/A-18 high AOA simulation. Although one of the high AOA piloted simulations (based upon 16% model data) predicted a low yaw rate spin mode approximately 1 year prior to the accident, there was not enough confidence in the data base to lower the automatic spin recovery mode engagement yaw rate threshold. This reluctance may have been due to the excellent departure and spin resistance exhibited by the F/A-18 up to that point in time and because of uncertainty in the aerodynamic data base.

FINAL CONCEPTS

FCS Spin Recovery Mode: As discussed earlier, the F/A-18 FCS was to have only an automatic reversion to spin recovery mode capability. However, as previously noted a cockpit mounted manual spin recovery mode switch was incorporated as a result of the spin accident in November 1980. Evolution of spin recovery mode logic for both the manual and automatic SRM is summarized next.

Manual SRM Logic: Manual SRM engagement/disengagement threshold/logic is summarized in figure 25.

- **ENGAGE LOGIC — SWITCH ON AND <120 KIAS**
- **DISENGAGE LOGIC — SWITCH OFF OR >250 KIAS**

Figure 25 Manual SRM Logic

During the design of the manual SRM, there was a great deal of effort made to insure that the air data system failures previously experienced during FSD flying qualities tests due to sideslip at high airspeed not occur during spin testing. This was necessary because reversion to the fixed gains mode of the FCS would prevent manual engagement of the SRM. This air data failure logic was selected because engagement of spin logic at high dynamic pressure would result in unacceptably sensitive and possibly dangerous aircraft response to pilot control inputs. One of the initial assumptions made was that, during a spin, the total dynamic pressure would be so low that dynamic pressure differences between left and right pitot-static probes could not exceed the preset failure monitor thresholds. However, during asymmetric load spin testing this assumption proved to be incorrect. During a 12,000 ft-lb asymmetric load spin, the difference in left and right Q_{C_i} exceeded the preset failure (ΔQ_{C_i}) thresholds, thereby causing reversion to fixed gains and preventing spin recovery until the failure cleared and the FCS was manually reset. Subsequent analysis revealed that, during the spin, the magnitude of sideslip was such that one of the L-probe pitot-static heads was registering negative dynamic pressure. As a result, the large ΔQ_{C_i} indication caused an air data failure and prevented access to the manual SRM. The air data system failure monitoring logic was subsequently changed so that only positive Q_{C_i} pressure indications are used to compute ΔQ_{C_i} mismatch values, and reasonable range thresholds were expanded to allow negative values. Also, both left and right Q_{C_i} values must fail reasonable range checks before a failure is declared. Excessive differences between left and right dynamic pressure signals will generate a pilot caution but are no longer used to declare an air data failure. During asymmetric load spin testing, reversion to fixed gains also occurred because of larger than expected values of left and right L-probe static pressures due to large sideslip. Left and right static pressure lines are now joined to eliminate that failure mode.

Auto-SRM Logic: As described earlier, the original SRM logic automatically reverted to the SRM if yaw rate exceeded 35 degree/second for ≥5 seconds. Automatic reversion to the CAS mode occurred if yaw rate decreased to less than 15 degree/second. As previously noted, engage/disengage yaw rate thresholds were subsequently changed to 50 and 30 degree/second, respectively, to increase departure/spin resistance and to preclude inadvertent SRM engagement during aggressive maneuvering. The need to further modify automatic SRM logic was recognized because of the low yaw rate spin mode identified during spin tests subsequent to the November 1980 spin accident. The ultimate goal was to replace the manual spin recovery mode switch with automatic SRM logic. Current automatic SRM logic is outlined in figure 26. This system has been extensively tested and has provided excellent spin recovery capability. However, additional testing is required to perfect the system to the point at which the manual SRM switch can be removed. A unique feature of current automatic SRM logic is that it provides full antispin control authority only if lateral stick is moved in the correct direction. The FCS automatically reverts/fades back to the CAS mode if prospin lateral stick is applied.

- **ENGAGE LOGIC — AIRSPEED <120 KIAS AND FILTERED YAW RATE X YAW RATE ≥225 (λ=7.2 SEC) AND CORRECT ANTI-SPIN LATERAL STICK APPLIED**
- **DISENGAGE LOGIC — AIRSPEED >250 KIAS OR FILTERED YAW RATE X YAW RATE <225 (λ=3.2 SEC) OR YAW RATE = 0 OR INCORRECT PRO-SPIN LATERAL STICK APPLIED**

Figure 26 Current Automatic SRM Logic

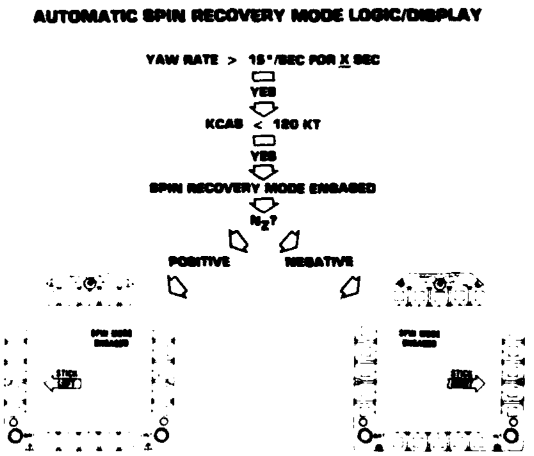


Figure 27 Spin Recovery Mode Display

Spin Recovery Displays: Both manual and automatic SRM logic also provide the pilot with spin recovery mode status information on the cockpit digital display indicators (see figure 27). These displays are specifically designed to provide the pilot with the information he needs to achieve spin recovery. Display information is programmed to appear on both of the cockpit Digital Display Indicators (DDI's). The DDI's are located on the right and left upper portions of the instrument panel. Normally, the DDI's are used to display weapon system information (radar, FLIR, Store Management, etc). However, in an out of control situation, the spin recovery displays have maximum priority and are automatically substituted in place of other display information which may be present. The spin recovery displays are designed to provide the pilot with two essential pieces of information: (a) SRM engage/disengage status and (b) antispin flight control instructions. These displays have tremendous potential for the F/A-18 and for future aircraft designs in that they allow the pilot to make an optimum spin recovery without having to determine the type or direction of spin or to recall the correct antispin controls. One significant problem associated with SRM display information that has been identified occurs during highly oscillatory spins. During intermediate yaw rate oscillatory spins, the F/A-18 will on occasion roll 360 degrees while continuing to spin in the established direction. During the roll, normal acceleration and body axis yaw rate change sign and the spin recovery display arrows momentarily change direction and point in the wrong direction. Review of test data indicates that momentary (i.e., less than 1 second) incorrect spin recovery arrow direction occurs when the sign of raw yaw rate or raw normal acceleration is opposite to that of their respective filtered values (filtered values are used in SRM logic). However, it is important to emphasize that the automatic SRM has operated satisfactorily during spin tests of the F/A-18 and, as such, serious consideration should be given to employing this concept in future advanced tactical aircraft.

HIGH AOA/AIR DATA SYSTEM FAILURE MONITORING

AOA System: The F/A-18 has AOA probes located symmetrically on each side of the forward fuselage as illustrated in figure 20. AOA information is utilized by both flight control and mission computers as illustrated in figure 28.

The FCS converts the two signals from each probe (total of four electrical signals) to digital quantities and through a voting logic computes the AOA signal to be used for control law computations. A correction factor is applied to the local AOA signal to give an approximation to true AOA to be used in control law computations. The range of the AOA probes is -14 to +56 degrees indicated AOA. Above 37 degrees true AOA, the INS is used to compute AOA's up to 90 degrees. This information is displayed to the pilot on the HUD.

AOA Failure Monitoring: AOA failure monitoring logic changed considerably during FSD testing. Evolution of AOA failure detection logic is briefly summarized in table III. Mismatch tolerances between left and right AOA probes progressively increased to prevent nuisance reversions of the FCS to the "fixed gains" mode. In the fixed gains, LEF's and TEF's are locked in position at time of failure while the FCS remains in the CAS mode. One of the first surprises of the FSD high alpha program occurred during departure resistance testing. During a departure at high subsonic Mach number, unexpectedly large mismatches in left and right AOA vane indications were of sufficient magnitude and duration to cause reversion of the FCS to fixed gains. Upon reverting to fixed gains, the leading edge flaps which should have locked in position at the time of failure, instead retracted, with the airplane in an unrecoverable poststall gyration. Recovery was delayed until the AOA failure cleared and the pilot manually reset the FCS to CAS. Revised control laws subsequently increased the AOA failure detection magnitude and duration thresholds and changed LEF control mechanization to prevent LEF'S/TEF'S from being driven to unusual/off schedule positions. An anto-reset FCS capability was also incorporated in the FCS together with inhibiting of AOA failure monitoring at low airspeed.

Air Data System: The FCS contains two dedicated dual channel pneumatic pressure sensors called Backup Air Data Sensor Assemblies (BADSA's). The filtered, uncalibrated sensor outputs are used for all inner loop control law gain scheduling. Pneumatic inputs are supplied to the BADSA's and other aircraft systems by two L-shaped pitot-static probes that are located symmetrically on the lower forward fuselage. Each L-probe has two static and one pitot pressure output. Each channel of the BADSA contains an absolute pressure transducer which measures static pressure and a differential pressure transducer which measures dynamic pressure (pitot pressure-static pressure). The air data system configuration is illustrated in figure 29.

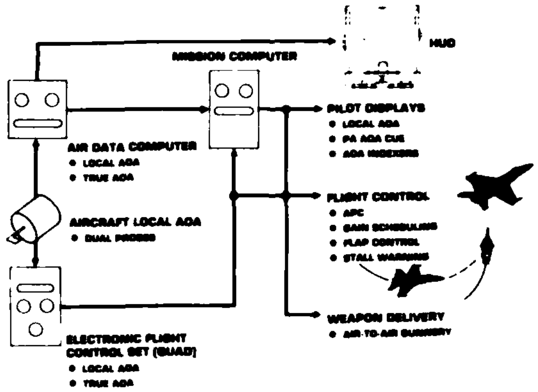


Figure 28 AOA System Configuration

Table III

AOA Failure Monitoring Evolution

Tolerance/Threshold AOA Mismatch (deg)	Tolerance/Threshold Time Constant	Remarks(1)	Control Law Version
3.5	0.5	Original tolerance/threshold.	3.11
5.0	1.0	Precludes FCS reversions to fixed gains due to sideslip.	3.12
15.0	1.0	Precludes FCS reversions to fixed gains due to sideslip.	3.21
15.0 30.0	10.0 10.0	AOA 15 deg AOA 45 deg	4.3
15.0	5.0	Auto-reset of failure if AOA in reasonable range. AOA failure monitoring inhibited at less than 100 kt.	6.X-8.X

NOTE: (1) In "fixed gains" the FCS remains in the CAS mode with nominal gains used by the flight control laws.

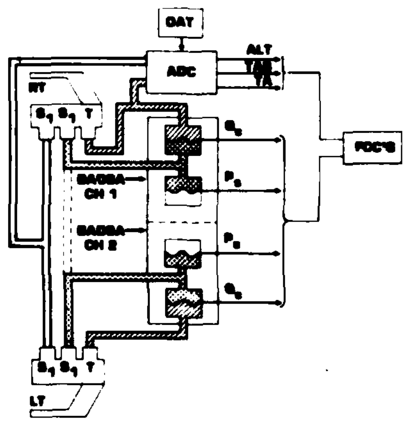


Figure 29 Air Data System

Air Data Failure Monitoring: With current control laws, an air data failure is declared only if both BADSA's fail reasonable range checks. If both BADSA's are in range, then the data from the BADSA with the highest dynamic pressure is used by the control laws. When an air data failure occurs, the FCS reverts to the fixed gains mode. During contractor FSD asymmetric stores high AOA spin testing, reversions to fixed gains occurred as a result of unexpectedly large sustained sideslip excursions which caused excessive differences between left and right static and dynamic pressure signals. These results had a significant impact on spin recovery capability as discussed in a previous section of this paper (see Manual SRM Logic).

HIGH AOA FCS DESIGN LIMITATIONS

One design area in which F/A-18 FCS design limitations become evident occurs at high AOA where nosedown pitch restoring moment requirements must be met. In March 1980, the wing leading edge extension (LEX) slots were closed to improve supersonic performance capability. However, closure of the LEX slots had a significant adverse affect on pitch restoring moment capability at approximately 50 degrees AOA. The zero pitch restoring moment CG was shifted approximately 5% forward (equivalent to approximately 0.1 ΔC_M) as illustrated by figure 30. This has caused delayed recoveries from high AOA flight conditions or AOA hang-up as illustrated in figure 24. Also noted in figure 30 are apparent scale effect differences on pitch restoring moment wind tunnel predictions between 6 and 16% models. Full scale development testing indicated that delayed recoveries from nose high - low airspeed flight conditions and from oscillatory spins were being occasionally encountered. At aft CG's and in the 50 to 60 degree AOA region, the airplane did not always respond immediately to neutral longitudinal controls or application of full forward stick. Subsequent flight testing was performed to identify the CG's at which these conditions (i.e., AOA hang-up) occurred for several external store loadings. The effect of aft CG's in the F/A-18 on AOA hang-up recovery is illustrated in figure 31, which shows that under these conditions full trailing edge down stabilator may not be sufficient to recover from a high AOA condition. As expected, the addition of external wing stores further reduced recovery capability by shifting the zero pitch restoring moment CG as much as 4% forward as compared with the fighter escort loading (wing tip and fuselage missiles only). Interim solutions toward avoiding AOA hang-up flight conditions include modified fuel sequencing to shift the CG forward (approximately 2%) and imposition of AOA/Mach number and CG placards to keep the airplane from entering the AOA hang-up region and to assure adequate recovery control power should AOA hang-up flight conditions be inadvertently encountered. A promising aerodynamic modification that has been designed by McDonnell Douglas and evaluated by NASA Langley Research Center involves a reduction in area of the wing leading edge extension as illustrated in figure 32. Wind tunnel test results (16% scale model) indicate that, in the AOA hang-up region, a 5% aft shift in CG position for zero pitch restoring moment would be realized with the modified LEX as illustrated in figure 32.

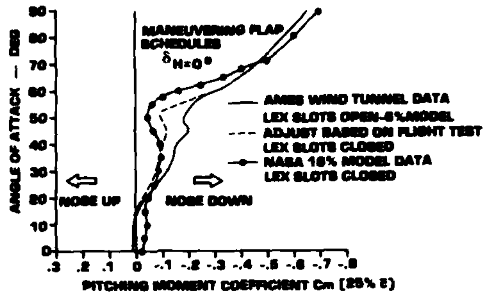


Figure 30 High AOA LEX Slot Effects

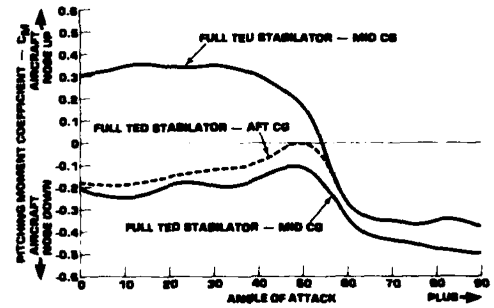


Figure 31 Aft CG AOA Hang-Up Recovery Effect

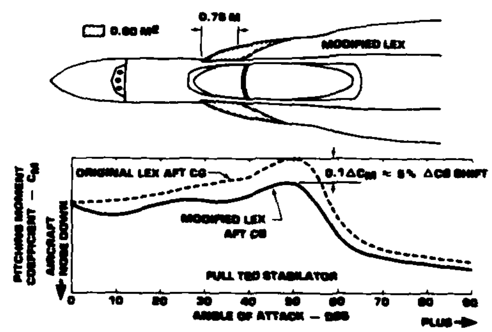


Figure 32 Proposed LEX Modification

FCS design changes can have significant limitations with regard to correcting flying qualities deficiencies particularly at high AOA. In some cases a relatively small aerodynamic design change may be a better and possibly the only solution to a flying qualities deficiency as opposed to a FCS design change for the same purpose.

CONCLUSION

The development of the F/A-18 control laws and specialized displays has been a major stepping stone in demonstrating the powerful flexibility and versatility of a digital FCS as an integral part of a total avionics system. The digital design and interface of the FCS with the MC and pilot displays has been instrumental in improving F/A-18 overall handling qualities, suppressing undesirable structural oscillations, developing high AOA departure resistance and spin recovery control laws, and in developing the associated unique pilot displays which have significantly reduced pilot workload and assisted in spin recovery. The flexibility and versatility inherent in the F/A-18 FCS design has permitted the implementation of unique methods to practicably effect required changes and to refine and develop these changes in a real-time flight test environment. The digital design and integration of the FCS into the overall avionics package of a given aircraft has and will be continually expanded in present and future generation aircraft. Testing of these new systems can be greatly assisted by utilizing their inherent flexibility throughout the flight test program

ACKNOWLEDGEMENTS

Portions of the information and data presented in this paper were obtained from McDonnell Douglas engineering personnel. In particular, special thanks to Messrs. E. H. Anderson, R. Anderson, H. Harschburger, L. Trame, and R. Pelican.

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LESSONS LEARNED IN THE DEVELOPMENT OF THE F-16 FLIGHT CONTROL SYSTEM

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SUMMARY

Flight control systems engineers have always been pioneers in design to achieve "survivability after failure". Historically, this capability has been accomplished by duplicating the functions of the flight control system and employing redundancy management to detect failures and to reconfigure for safe operation within an acceptable probability. Protection against failure by redundancy always results in a design trade between reliability (mean time between failure) and safety (mean time between castrophic loss). Fortunately, improvements in hardware reliability have kept pace with increased system complexity, and redundant systems have become the norm in all modern flight control systems on high-technology military aircraft.

The Fort Worth Division of General Dynamics has been heavily involved with failure-tolerant flight control systems for almost 20 years through production of the F-111 and the F-16. The need for survivability after failure is most obvious on the F-16, which has the world's first production fly-by-wire flight control system. Fly-by-wire is an absolute necessity on the F-16 because the aircraft was designed to be statically unstable and cannot be controlled without the artificial stability provided by the flight control system. The well over 250,000 flight hours on the F-16 have provided several lessons that are not readily apparent on programs not in a production environment. The analytical system failure probabilities predicted by traditional analysis for random failures have been essentially proven. However, one fallacy in this traditional analysis must still be overcome. Because of difficulties in analytical definition, several external factors may not be included in the failure analysis of flight control systems. These factors, although not directly a part of the flight control system, can render redundancy useless if not considered. Examples of these external factors are pilot interface, ground maintenance, structural resonance, environmental conditions, indirect electrical hazards, and other system failures. These factors are not unique to the F-16 but are common to all aircraft with fly-by-wire flight control systems. This paper discusses examples of how several of these factors manifested themselves in the development of the F-16 and how the F-16 flight control system has evolved to minimize their effect. One of the most significant evolutions to aid in the isolation and resolution of problems is the time sequenced data provided by the F-16 maintenance memory.

GENERAL DESCRIPTION OF THE F-16 FLIGHT CONTROL SYSTEM

The F-16 fly-by-wire Flight Control System (FLCS) consists of many interacting components, as illustrated in Figure 1. Pilot commands to the FLCS are generated by a sidestick controller, rudder pedals, and switches in the flight control panels. The electrical commands from these cockpit components are fed to the Flight Control Computer (FLCC), which can be viewed as the "heart" of the FLCS. The FLCC also receives aircraft inertial motion inputs from pitch, roll, and yaw gyros and from normal and lateral accelerometers. Additionally, air data information is transmitted to the FLCC after measurement by total pressure, static pressure, and angle-of-attack (AOA) probes; conversion of pneumatic pressure to electrical signals by the Pneumatic Sensor Assembly (PSA); and signal conditioning by the Electronic Component Assembly (ECA). The FLCC combines these inputs through appropriate control laws to produce control-surface commands that position five primary control surfaces through Integrated Servoactuators (ISAs). Primary control surfaces are defined as two flaperons, two all-moveable horizontal tails, and a rudder. The FLCS provides information back to the pilot through various indicator, caution, and warning lights located in the cockpit.

The F-16 also incorporates several secondary control surfaces - the leading-edge flap and speed brakes. The leading-edge flap is driven by a power drive unit that is programmed by the ECA as a function of AOA, altitude, and Mach number to provide optimum wing contour during maneuvering. The two speed brakes are directly controlled by the pilot.

THE FLY-BY-WIRE DECISION

From the very beginning, General Dynamics decided to take advantage of the reduced drag and lower weight provided by designing the F-16 to be statically unstable at subsonic speeds (up to 10 percent MAC for low angles of attack). With this decision came the need for artificial stability provided by the FLCS. The FLCS must be designed with high reliability and fault tolerance because flight characteristics are such that the pilot could not fly the F-16 without artificial stability. Since aircraft safety is totally dependent on the electronic artificial stability system, the need for mechanical linkage is eliminated. Pilot commands can be electrically fed into the FLCS like the other critical inputs.

PROTECTION AGAINST FLCS FAILURE

The F-16, like most modern high-performance aricraft, is protected against FLCS failure in three ways:

1. High reliability
2. Redundancy
3. Built-in Test.

To protect against system failures, it is fundamental to make the mean time between failure (MTBF) as long as possible through high reliability of system components. However, for many cases in complex flight control systems, component reliability cannot be made high enough to eliminate the need for system tolerance of component failure by some sort of failure detection and correction scheme. On the F-16, protection against failure is achieved by redundancy. Critical functions are replicated up to four times in the FLCS. In the event of consecutive component failures, the system can sense the error by comparison of redundant branches and then provide a valid output command by a selection process.

The F-16 FLCS also employs built-in self test to protect against failure. Inherent in the design of the redundancy management system is the assumption that it is working properly. One of the primary functions of built-in self test is to check the capability of the redundancy management system to detect and correct failures. Of course, self test is also designed to detect failures in the redundant branches prior to flight.

THE F-16 REDUNDANCY MANAGEMENT SYSTEM

Redundancy level is a term that is used in failure-tolerant systems as an indication of the amount of failure protection provided. A redundant system is one where various system functions are replicated so that failures in the system can be detected and corrected by use of alternate paths.

The level of failure protection for redundant systems is classified by the number of like failures that can be sustained without loss of function. Fail-operational systems can withstand one failure. Two-fail-operational systems can stand two consecutive like failures.

Another term used relative to failure protection is fail-safe. A fail-safe system suffers degradation on failure, but the degradation does not prevent safe operation. In an aircraft flight control system fail-safe means the pilot can return the aircraft and land safely.

Many factors influenced the redundancy level and failure protection selected for the F-16. Two of the most important factors to be remembered about redundancy are that

1. Redundancy (failure protection) reduces the chance of catastrophic failure.
2. Redundancy decreases reliability, may increase the probability of takeoff aborts, and increases costs.

The relative importance of these two factors is a driving influence in the selection of the proper protection level.

The selection of redundancy and failure protection is influenced by other factors in addition to safety probability, reliability, and cost. Some of the more significant factors are

- o Emotional acceptability by system users based on previous experience with other systems.
- o MTBF of mature versus new hardware.

- o Physical geometry of the airplane.
- o Interface compatibility with other parts of the system.

On the YF-16 and F-16 fly-by-wire FLCS, the redundancy level varies from none to quadruple, and failure protection varies from none to two-fail-operative/fail-safe. The interrelationships among the F-16 redundancy and failure protection levels are shown in Figure 2.

The highest failure protection used on the F-16 is in the quadruple-redundant flight control computer (FLCC). This redundancy level provides two-fail-operative protection in the pitch axis and two-fail-operative/fail-safe protection in the roll and yaw axes. One of the most significant factors considered on the F-16 was the accepted failure protection levels on other high-performance aircraft flying in 1972. The F-111 is typical of these aircraft. The FCS on the F-111 is a single-fail-operational electronic system in conjunction with a full-time mechanical system. The logical reaction to removing the mechanical system on the F-16 was to increase the electrical failure protection to two-fail-operational. This increased protection could be obtained for a relatively small change in electrical complexity. A triple-redundant implementation was necessary for a single-fail-operational system. By addition of only one more branch, another entire level of failure protection could be obtained.

Figure 3 is a schematic of the F-16 pitch axis, which shows the system redundancy levels. The pitch axis is the most critical on the F-16 because of the longitudinal static instability of the aircraft. This figure shows two of the most critical parts of the redundancy management system of the F-16 FLCS. The first is the signal "selector". This device selects a good electronic channel even after two consecutive electronic branch failures. The evolution of the selector design was a significant part of the development of the F-16 analog fly-by-wire FLCS. The second critical part of the redundancy management system is the electrical-to-mechanical interface provided by the servoamplifier monitor system and the self-contained hydro-mechanical failure detection and correction logic in the ISA.

FLIGHT EXPERIENCE

The F-16 now has accumulated well over 250,000 flight hours with eight different Air Forces. This experience has provided the basis for many "lessons learned". The first observation is that the FLCS has done an excellent job of protecting against the typical sequential failures it was designed to protect against. This observation is of no particular surprise, since the FLCS was designed for a safety probability of millions of flight hours and, although FLCS flight experience is considerable, it is nowhere near the predicted safety probability.

Experience has shown that the biggest concerns relative to aircraft fly-by-wire flight control systems are unforeseen factors that can defeat redundancy. Examples of these factors are

- o Pilot Interface
- o Ground Maintenance
- o Structural Resonance
- o Other System Failures
- o Indirect Electrical Hazards
- o Environmental Conditions.

Protection against factors that can defeat redundancy is two-fold. First, experience from previous flight control system designs is woven into the basic design of a new system from the beginning. However, if the new system is an advancement of technology, there will probably be unforeseen factors. Therefore, the FLCS design and operations team must be very responsive to indications of any problem during flight operations.

Usually a potential problem will give a warning if the proper personnel are aware of the symptoms. An experienced failure analysis team is necessary to recognize potential problems. Many potential problems will not manifest themselves until the aircraft moves out of the development phase into a general operational environment. In summary, there is no substitute for flight experience in operational usage.

On the F-16, the experience gained in flight testing and operational usage has been used to improve the FLCS. Illustrative examples based on experience are as follows:

- o Maintenance Memory
- o Structural Resonance
- o Ground Maintenance
- o Pilot Command.

Maintenance Memory

Early in the F-16 program it became apparent that additional time-sequenced data would be a big aid in isolating and resolving system problems. Pilot statements of system problems were not always sufficient to identify the problem source. The pilot is not always aware of failure symptom sequence or is not able to remember the sequence accurately after flight. This problem was aggravated by the fact that some failures give multiple indications that cannot all be observed in a short time interval. In some cases, the pilot is not always aware of any procedural errors he may have made.

The early F-16 had hardware failure annunciators; however, these devices did not always provide adequate information to resolve the cause of transient problems that no longer existed after flight. The need for better failure data resulted in the development of a maintenance memory for the F-16 Flight Control System.

Figure 4 shows the various components that make up the F-16 maintenance memory system. A non-volatile microprocessor-controlled memory was installed in both the ECA and FLCC. The data stored in these two memories were also recorded into a non-volatile memory located on the ejection seat. The data stored in these memories can be retrieved from the ECA, FLCC, or seat data recorder by ground test equipment. Table 1 describes the features of the FLCS maintenance memory.

A new piece of ground equipment was developed to allow maintenance personnel to interrogate the memory directly without removing any FLCS equipment. This FLCS "fault finder" can be plugged into the flight control panel in the cockpit.

The F-16 FLCS maintenance memory is not a "crash recorder" but does have many features that make it very valuable. The seat unit was designed to survive the ground impact of an ejected seat. It has also been shown capable of surviving most ground impacts even if it is still installed in the aircraft. However, the seat unit has very limited protection against fire. In many cases data can also be retrieved directly from the FLCC and ECA memory chips, even when the units are almost completely destroyed.

Structural Resonance

When firing the 20mm gun on early production aircraft, yaw failure indications and yaw transients occurred occasionally. These symptoms were very perplexing since no problems had been experienced during the FSD program, all FLCS units were qualified to the gun-firing environment, and gun-firing frequencies are well above the response capability of the F-16 analog flight control system.

The cause of the problem was finally traced to a local resonance of the structure that held the lateral accelerometer, which is positioned in the F-16 as shown in Figure 5. The failure scenario was as follows:

- Gunfiring introduces high-energy oscillations into the airframe at 100 Hz.
- Local resonance of lateral-accelerometer mounting structure occurs at 300 Hz.
- Lateral acceleration due to the third harmonic of gun-fire frequency is amplified several orders of magnitude by local structural resonance.
- Frequency and acceleration levels act on the mechanical components of the torque-balance accelerometer in a non-linear manner to produce a DC off-set.
- The erroneous lateral acceleration signal causes the Stability Augmentation System (SAS) to command rudder motion.
- Rudder motion causes a yaw transient.
- Channel differences produce failure indications.

One of the first questions that comes to mind is why this problem was not seen during FSD testing. Further investigation showed that a small flight test instrument was installed on the same mounting structure as the accelerometer unit. This added equipment changed the natural frequency of local structure so that resonance did not occur with harmonics of the gun-fire oscillations.

The problem was resolved by re-analysis and test of the mounting structure. An isolation mount was then desired for the accelerometer package to essentially eliminate gunfiring frequencies (see Figure 6).

The lessons learned from this experience were as follows:

- Detailed analysis and testing is required of local structure near flight control sensors.
- Seemingly small equipment or mounting changes can have a big effect near FLCS sensors and must be carefully monitored.

Ground Maintenance

A typical symptom of a problem in the ground maintenance category would be aircraft pitch transients while flying in icing conditions. Problems with icing were not considered highly probable because all five FLCS air data probes have separate anti-ice heater power circuits from the essential aircraft bus; before every flight the pilot turns on probe heat or, if he forgets, it is automatically turned on at takeoff.

The cause of this problem was isolated to the maintenance personnel having left all five probe heater circuit breakers open. The failure scenario was as follows:

- Ground maintenance man leaves all probe heat circuit breakers open.
- Probes ice up in flight.
- Ice causes random motion to occur on at least two angle-of-attack probes.
- Angle-of-attack feedback to FLCS produces pitch transient.

Several questions come to mind after studying this problem scenario. First, one wonders why a maintenance man would open all five probe heat breakers. The reason is that probe heat is a burn hazard to maintenance personnel when performing tasks that require aircraft weight-on-wheels logic to be bypassed (e.g., gear tests, FLCS tests, etc). Also, energizing probe heat on the ground can reduce probe heater life and can cause significant damage if ground protective covers are not removed.

The second question deals with the reasons that ground procedures did not catch this problem. All ground maintenance and pre-flight procedures call for verification that probe-heat breakers are energized. Apparently, the frequency with which these breakers are opened coupled with unknown situations result in these breakers being overlooked. The problem has been resolved by multiple pre-flight verification of probe heat, and a proposal has been submitted for probe heat to be included in the built-in test conducted by the pilot.

The important lessons learned from this experience are that

- Where a single maintenance man must routinely disable critical redundant circuits, the chances are high that at some time the circuits will be left disabled for flight, thus defeating redundancy.
- Verification of probe heat should be included in future FLCS monitor and self-test designs.

Pilot Command

A typical symptom of an unforeseen factor in the pilot command category would be departure of the aircraft from controlled flight while avoiding defensive aircraft during a ground attack mission. The F-16 is essentially "eyes out of the cockpit" for air-to-air combat, with the flight control system automatically allowing the pilot to obtain maximum acceleration and AOA without watching his instruments. However, in early F-16 versions, the pilot had to observe certain roll-rate, angle-of-attack, and acceleration restrictions for air-to-ground loadings.

The cause of this type of problem was rooted in the fact that pilots had become extremely comfortable with the ability of the flight control system to protect them from over-command in air-to-air maneuvering so that, when carrying out air-to-ground missions, they forgot and exceeded roll-rate-command limitations.

Discussions revolving around this typical symptom of pilot over-command always brought out the statement that a qualified pilot should always be aware of his situation and observe all limitations, as is required on conventional aircraft. However, further investigation revealed that the ability in air combat to maneuver with "reckless abondon", with eyes out of the cockpit, has been one of the most outstanding features of the F-16. Pilots have come to depend on the ability of the FLCS to maximize his commands, thus allowing him to concentrate on other aspects of his mission. Because of the success of this concept in air combat, the pilot has difficulty re-adjusting to conventional limitation for air-to-ground loadings.

Resolution of this problem was obtained by implementing a system to maximize pilot commands for air-to-ground loadings. A pilot-selectable switch allows the pilot to switch between air-to-air and air-to-ground maximizer configurations. A caution light tells the pilot if he is in the wrong switch position for the existing aircraft loading.

The primary lesson learned from this experience is that if an FLCS feature is so successful that pilot reliance upon it becomes "second nature", he may not always revert to conventional operation in other modes.

SUMMARY OF LESSONS LEARNED

The lessons learned from flight experience discussed in this paper are summarized as follows:

- The biggest concerns will occur as a result of unexpected conditions.
- Operational flight is the only way some of these problems will be identified and rectified.
- A modern fly-by-wire FLCS needs some type of maintenance memory to record time-sequenced data for identification and resolution of problems.
- Local structure can cause problems and therefore requires special attention.
- Electrical power deserves more attention than industry has given it in the past.
- During maintenance, critical FLCS circuits must routinely be disabled. Therefore, more than the "normal" safeguards must be observed to prevent flight unless the circuit is enabled again.
- If pilot reliance on an FLCS feature becomes "second nature", the cases where he must go back to conventional operation should be minimized.

CONCLUSIONS

The technical community spends a tremendous amount of effort on the theoretical analysis of flight-critical systems such as the FLCS. Detailed theoretical probability calculations are performed to determine the best system architectural approach to solve FLCS failure-protection problems. Usually these determinations are based on calculations showing millions of flight hours before system failure. However, experience has shown that the real concern is learning how to minimize the effect of unknown problems. The relationship between theoretical safety probabilities and susceptibility to unknown problems is not clear.

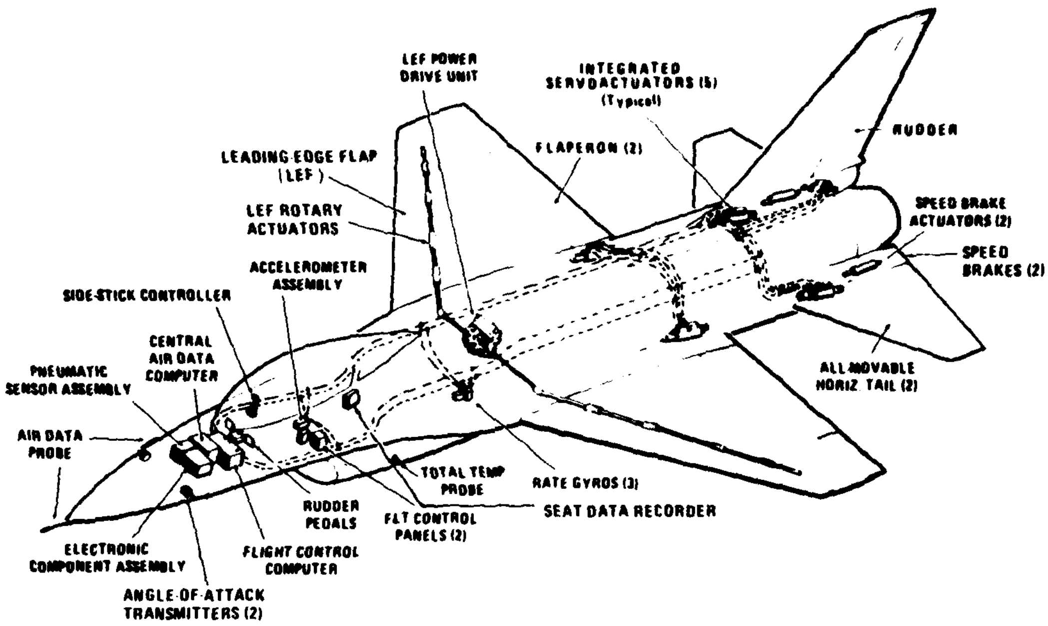


FIGURE 1 F-16 FLIGHT CONTROL SYSTEM COMPONENTS

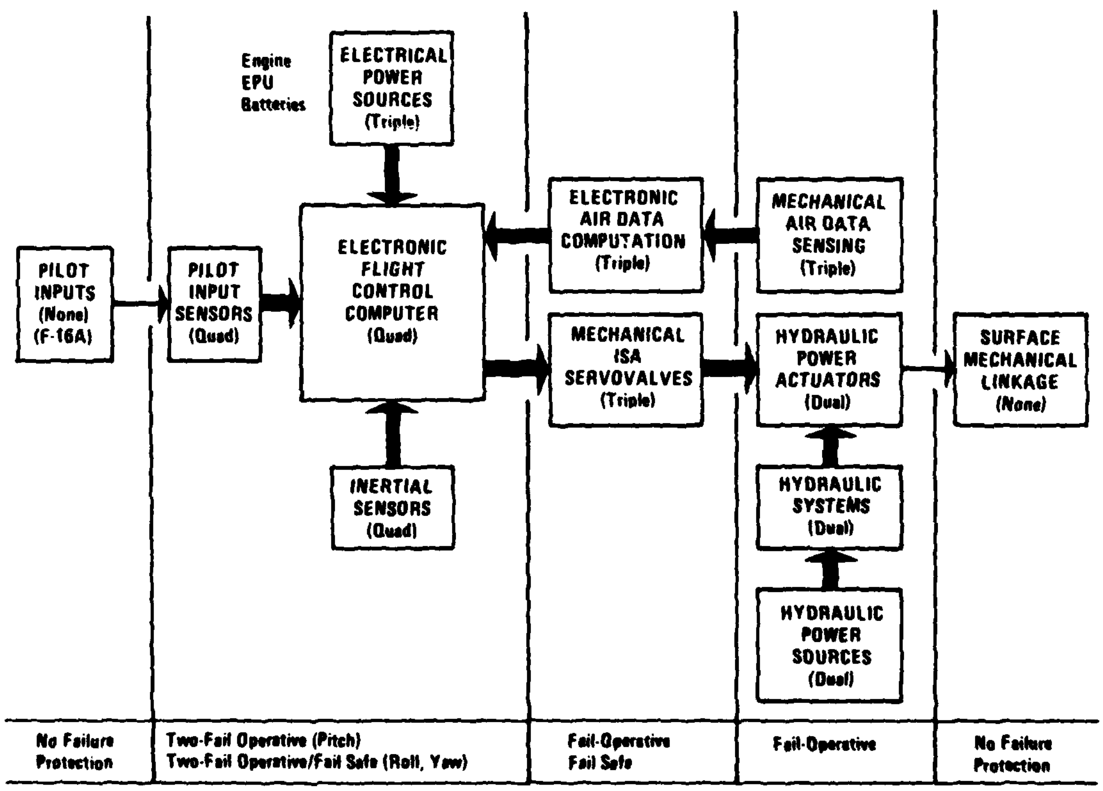
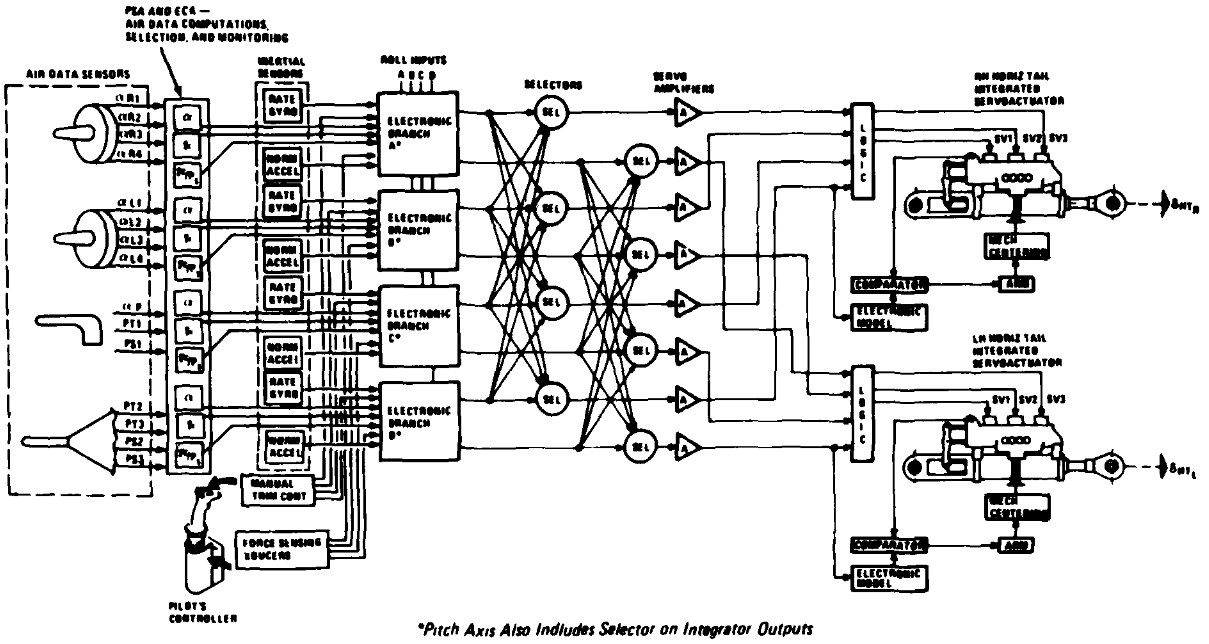


FIGURE 2 F-16 REDUNDANCY AND FAILURE PROTECTION LEVELS



*Pitch Axis Also Indludes Selector on Integrator Outputs

FIGURE 3 HOW ELECTRONIC REDUNDANCY WORKS
(PITCH AXIS SHOWN)

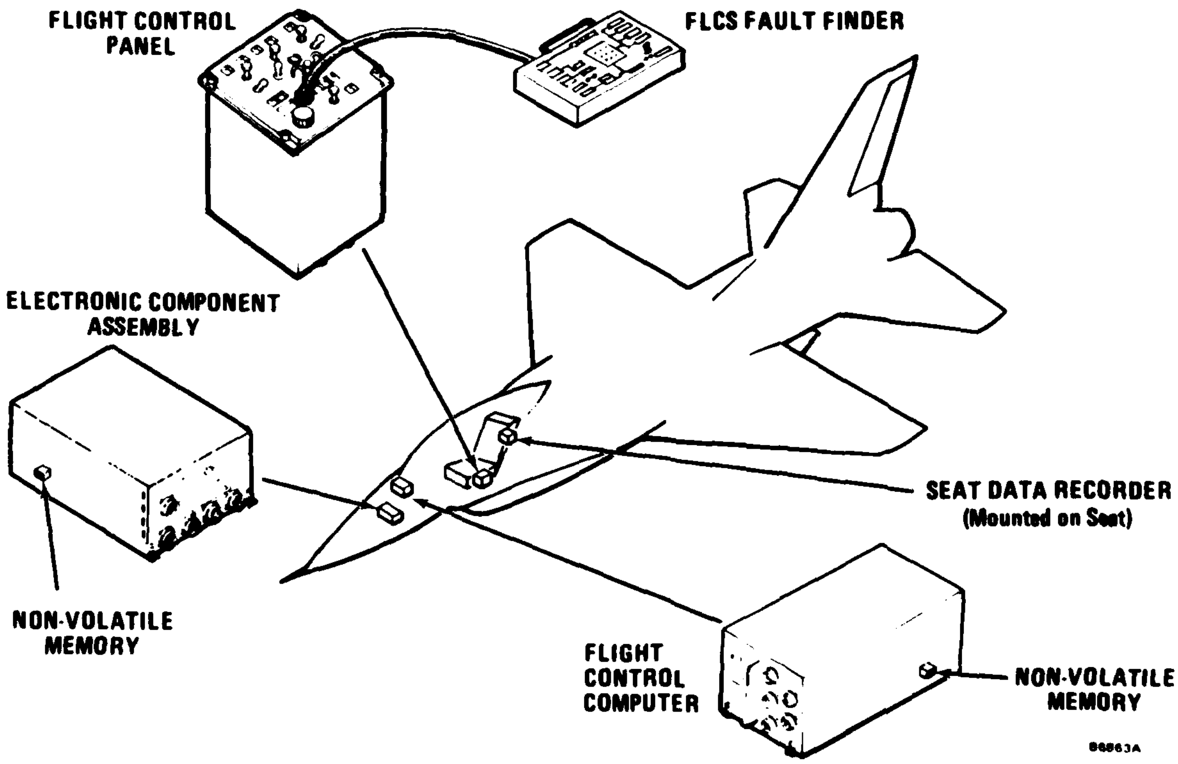


FIGURE 4 THE F-16 FLIGHT CONTROL SYSTEM MAINTENANCE MEMORY SYSTEM

TABLE 1 MAINTENANCE MEMORY FEATURES

- **RECORDED FUNCTIONS**
  - ✔ TIME AFTER TAKE-OFF
  - ✔ ALL REDUNDANCY MANAGEMENT MONITOR STATES
  - ✔ CRITICAL FLCS SWITCH POSITIONS
  - ✔ FLCS BATTERY DISCHARGE
  - ✔ ANGLE OF ATTACK
  - ✔ AIRSPEED
  - ✔ ALTITUDE
  - ✔ SELF-TEST SURFACE POSITION DISCRETES
  - ✔ FLCS CAUTION AND WARNING LIGHTS
- **RECORDING TIMES**
  - ✔ EVERY 64 SECONDS
  - ✔ EVERY TIME A REDUNDANCY MANAGEMENT STATE CHANGES, FLCS LIGHT ILLUMINATES, OR CRITICAL SWITCH IS THROWN
- **SIZE**
  - ✔ APPROXIMATELY 50 SQ. INCHES OF BOARD AREA IN ECA AND FLCC
  - ✔ 4" x 5" x 2" RECORDER MOUNTED ON EJECTION SEAT
- **MEMORY TECHNOLOGY**
  - ✔ N-MOS EA ROMS (4 1K By 4)

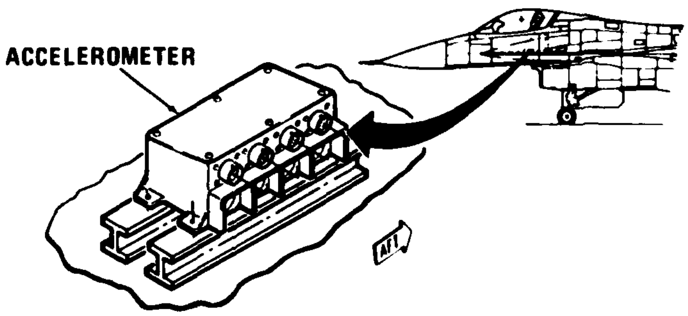


FIGURE 5 F-16 ACCELEROMETER ASSEMBLY

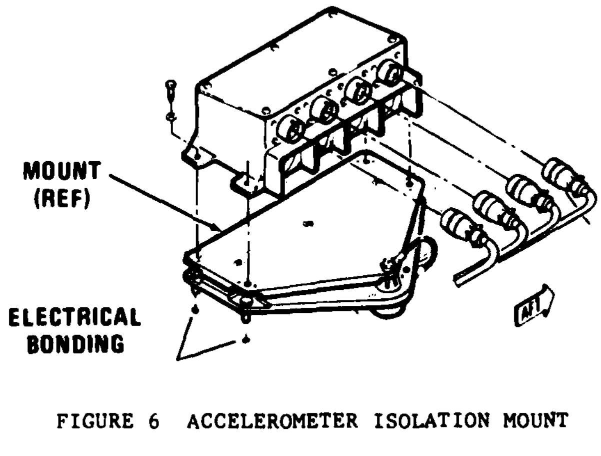


FIGURE 6 ACCELEROMETER ISOLATION MOUNT

MIRAGE 2000 : CDVE ET SECURITE

par

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RESUME

Le MIRAGE 2000 est réalisé pour présenter la meilleure efficacité en combat. Il utilise dans ce but des CDVE évoluées qui permettent une application raisonnable du Contrôle Automatique Généralisé de l'avion. Une recherche poussée de la sûreté du fonctionnement du système de pilotage nous conduit à un objectif de sécurité nettement supérieur à celui observé sur les avions antérieurs. L'ensemble de nos objectifs de performances et de sécurité ont fait l'objet de validations théoriques et pratiques en laboratoires et en vol, pour justifier une utilisation opérationnelle des CDVE.

INTRODUCTION

Le MIRAGE 2000 est le premier avion Européen, et le seul aujourd'hui, qui utilise opérationnellement des COMMANDES DE VOL ELECTRIQUES (C.D.V.E.) sans timonerie mécanique. Les CDVE de l'avion sont issues de l'expérience acquise par les équipes spécialisées des AVIONS MARCEL DASSAULT - BREGUET AVIATION depuis plus de trente ans sur toutes les commandes de vol de tous les avions de la famille MIRAGE ; elles associent l'efficacité opérationnelle et la sécurité en vol. La sécurité découle de la suppression des consignes et des restrictions d'emploi à observer par le pilote, en raison de l'architecture redondante du système des limitations automatiques sûres et de la gestion intégrée des pannes.

OBJECTIFS DE CONCEPTION

Nous avons décidé lors de la conception initiale du MIRAGE 2000 d'utiliser pleinement les avantages du traitement électronique dans les commandes de vol, en bénéficiant des enseignements des travaux réalisés pour l'ACF (1975) et de l'expérience des commandes de vol à transmissions électriques du MIRAGE III (1955) et du MIRAGE F1 (1968).

- sur le plan fonctionnel, les CDVE du MIRAGE 2000 sont définies pour répondre à trois objectifs principaux, qui ont tous pour but d'accroître l'efficacité de l'avion :
  - . Permettre le pilotage de l'avion instable, ce qui est bénéfique sur le plan des performances (vitesse d'approche, marge de manoeuvre, ...) et minimise les contraintes de chargement de l'avion ;
  - . Pratiquer opérationnellement les hautes incidences subsoniques, éventuellement jusqu'à vitesse nulle, ce qui est déterminant en combat ;
  - . supprimer les limitations en fonctionnement normal.
- sur le plan de la sécurité, les CDVE sont réalisées pour satisfaire deux objectifs majeurs :
  - . Obtenir un niveau de sécurité global supérieur ou égal à celui des avions précédents ;
  - . Obtenir une bonne protection contre les risques de guerre (vulnérabilité aux armes, à l'IEM...)

Le niveau global de sécurité que nous avons défini pour les CDVE et ses systèmes associés (hydraulique, électricité et anémométrie) résulte d'une étude AMD-BA conduite à partir de recueils d'incidents majeurs et d'accidents d'avions de combat dans le monde entier. La statistique établie repose sur plus de 10^7 heures de vol, effectuées en plus de 10 ans par plus de 5000 avions modernes Français et Etrangers ; elle conduit à un taux de pannes catastrophiques des commandes de vol qui est inférieur à la réalité par suite de l'imprécision de certaines enquêtes (pertes de contrôle non expliquées, avions non récupérés...).

Ce taux λ_c, hors fait de guerre, est compris entre $3{,}5 \times 10^{-6}$/h (280.000 h) et $5{,}6 \times 10^{-6}$/h (180.000 h). En réalité, nous pensons que ce taux est plus fort ; d'ailleurs les exigences définies en ce domaine pour les avions Américains sont fixées à 10^{-5}/h (100.000 h) dans la norme MIL 9490D. Dans le cas du MIRAGE 2000, qui bénéficie de toute notre expérience et d'un système complet réalisé par AMD-BA, nous avons retenu un objectif de sécurité ambitieux puisque d'un ordre de grandeur supérieur.

$$\boxed{\lambda_c \text{CDVE MIRAGE 2000} < 10^{-6}/h}$$

AUGMENTATION DE LA SECURITE GENERALE PAR LES CDVE

En fait les pannes catastrophiques de commandes de vol ne représentent qu'une part très faible dans les causes de pertes d'avion. Les accidents imputables à des fautes de pilotage sont beaucoup plus nombreux (5 à 10 fois). Les commandes de vol électriques du Mirage 2000, par l'amélioration des qualités de vol qu'elles apportent d'une part et par les protections automatiques qu'elles assurent d'autre part, doivent augmenter très sensiblement la sécurité de l'avion :

- les qualités de vol procurées par les commandes de vol sont telles que le pilote peut sans risque mettre ses commandes en butée en tous les points du domaine de vol. Il peut également, par des manoeuvres dans le plan vertical, amener l'avion au voisinage de la vitesse nulle, la récupération s'effectuant sans difficulté. La précision des commandes de vol repousse les risques de pompage piloté (PIO) et facilite les opérations délicates telles que le ravitaillement en vol.

- Les protections automatiques limitent le facteur de charge, les braquages de gouvernes et les vitesses de roulis afin d'empêcher le pilote de dépasser les charges structurales limites. Elles empêchent également le pilote d'atteindre des incidences qui pourraient poser des problèmes de contrôle tant que la vitesse est suffisante pour créer effectivement ces problèmes.

Enfin le pilote a la possibilité, sur la commande de profondeur, d'écraser une butée élastique, ce qui permet de dépasser la charge limite sans atteindre la charge de rupture, afin d'éviter le sol par exemple.

DEFINITION DES CDVE

Les CDVE du Mirage 2000 sont définies, réalisées et validées par les AMD-BA, comme pour tous les avions de la famille MIRAGE. Ceci permet d'assurer une unité de conception et une intégration poussée des fonctions mécaniques, hydrauliques et électroniques dans l'avion.

Les CDVE du MIRAGE 2000 sont caractérisées par les traits essentiels suivants :

. Utilisation de niveaux de redondances modulés suivant l'axe de pilotage, les fonctions réalisées, les risques de mode commun (foudre, IEM, impacts...) ;

. Reconfiguration fonctionnelle suivant l'état de dégradation des capteurs ;

. Adoption d'une signalisation globale, qui supprime les interprétations ;

. Utilisation d'un dispositif automatique de gestion des défaillances du système ;

. Utilisation d'une commande électrique d'ultime secours.

- Sur le plan de la conception générale, le système CDVE est présenté figure 1 où apparaissent :

  . deux circuits hydrauliques ;
  . quatre alimentations électriques ;
  . les capteurs pilote et inertiels à quatre voies en tangage ;
  . les capteurs anémo-barométriques triples liés à un calculateur à trois processeurs pour isolement des foudroiements ;
  . les Racks des chaînes CDVE ; quatre chaînes en tangage et trois chaînes en transversal.
  . les capteurs et le calculateur spécifique de la commande d'ultime secours ;
  . les servocommandes électro-hydrauliques à double corps et entrée électrique de secours indépendante ;
  . les becs.

- Les générations et distributions hydrauliques sont basées sur l'utilisation de deux circuits distincts à 280 bars (4000 PSI). Une pompe est entraînée par le réacteur, la seconde par l'intermédiaire du relais d'accessoires (figure - 2). Une électro-pompe primaire est branchée en parallèle sur le circuit 2 ; elle est alimentée par le réseau 28 V de l'avion.

  L'expérience des circuits hydrauliques de nos avions montre que la perte des deux circuits dans un même vol n'a jamais été rencontrée, et que la perte d'un circuit a un taux voisin de 5×10^{-5}/h.

- Les générations et distributions électriques sont présentées figure 3. Elles sont conçues pour alimenter deux chaines de CDVE en toutes circonstances, y compris l'extinction réacteur à très faible vitesse (en-dessous de 100 Kt).

  . Une génération est réalisée à partir de chaque circuit hydraulique. Les alimentations E_1 et E_2 correspondantes sont exclusivement réservées aux chaînes 1 et 2 des CDVE.

  . Une génération E_3 est liée aux générateurs principaux de l'avion. Elle alimente la chaîne 3 des CDVE, le calculateur triple lié aux capteurs anémo-barométriques et le calculateur de mécanismes des becs.

  . Une génération E_4 réalisée par la batterie de l'avion. La chaîne 4 des CDVE est alimentée par un convertisseur extérieur (DC/DC) alors que la chaîne de secours dispose d'un convertisseur autonome.
  Selon nos études, et nos essais, nous estimons extrêmement improbable d'avoir plus de deux alimentations électriques en panne, durant un vol. En outre, notre expérience montre qu'un foudroiement n'a jamais entraîné de destruction de batterie en bon état.

- L'architecture des CDVE résulte de la formule aérodynamique de l'avion, du nombre des gouvernes - quatre élevons en tangage et en roulis, un drapeau, les becs - et des effets transitoires admissibles lors de pannes.

  Après une étude et des essais détaillés, nous avons retenu deux dispositions de base et une redondance modulée :

  . Les calculateurs de CDVE sont au nombre de cinq :

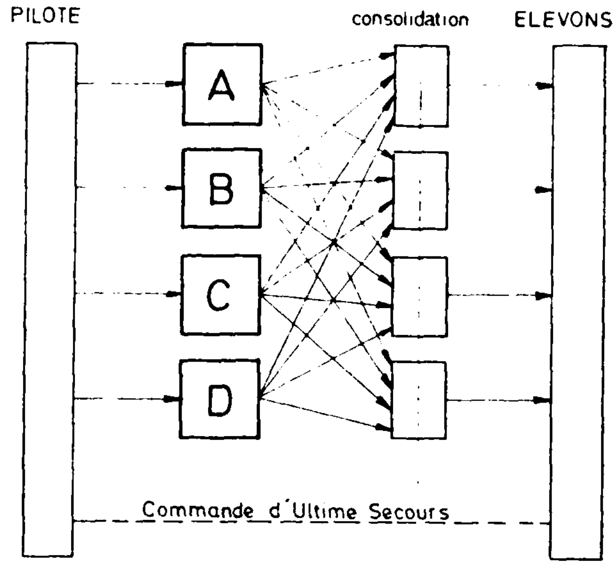
    * Deux Racks analogiques pour les quatre élevons et pour le gouvernail de direction.
    * Un Rack, numérique, pour les gains automatiques.
    * Un calculateur analogique dual pour les becs.
    * Un calculateur analogique pour la commande de secours.

    Cette disposition assure un niveau complémentaire de redondance vis à vis des pannes de mode commun (impact, feu, débranchement de calculateur ...).

  . Les chaînes de CDVE normales sont tronçonnées par des voteurs électroniques (figure 4 dans le cas de la profondeur). Ceci permet de multiplier les tolérances aux pannes et de limiter leurs effets tout en égalisant les tolérances de réalisation. On remarquera le doublement des voteurs d'ordres pour tenir compte de la panne interne des voteurs.

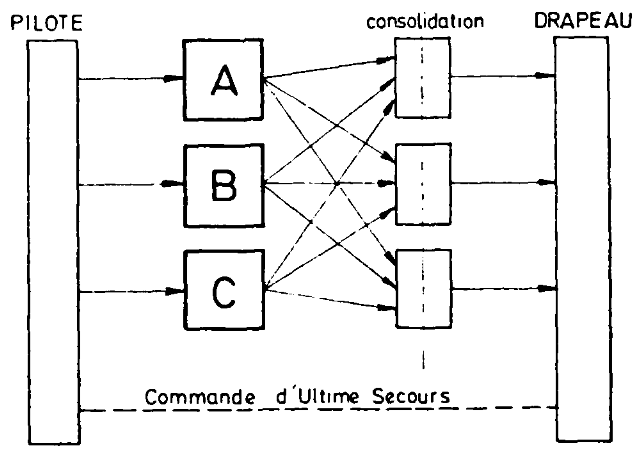
- Les chaînes quadruples sont utilisées en profondeur et en gauchissement ; elles sont liées aux quatre servocommandes des élevons et permettent de supporter toutes secondes pannes sans modifier les performances. La commande de secours constitue une redondance supplémentaire pour ces axes de pilotage.

La solution retenue peut être symbolisée par le diagramme suivant :



- Les chaînes triples ont été retenues quand l'avion peut être piloté en mode dégradé après la perte de certaines fonctions, ou lorsque l'on sait créer une fonction secours, même moins performante, dans d'autres calculateurs. Ce sont :

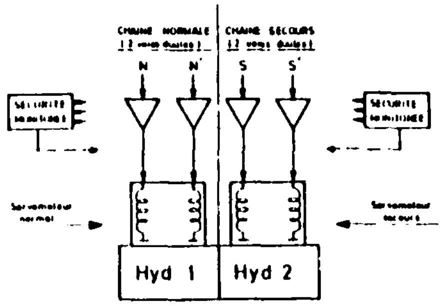
  . l'élaboration des ordres en direction, complétée par la commande de secours, que nous symbolisons ainsi :



  . les détections gyrométriques et accélérométriques transversales (p, r, Jy)
  . l'élaboration des gains automatiques et de l'incidence pour limitations automatiques.

  Ces fonctions sont présentées plus loin.

- Les chaînes doubles duales monitorées, sont retenues en aval des élaborations d'ordres, pour les cinq servo-commandes de gouvernes. Ce type de redondance est symbolisé de la manière suivante :



SECURITE DOUBLE DUALE MONITOREE

DOUBLE : car il existe un servo-moteur NORMAL et un servo-moteur SECOURS

DUALE : car chaque servo-valve comporte deux moteur couples. Un moteur couple est lié à une voie, le second à l'autre, dans une même chaîne (N-N', S-S')

MONITOREE : car chaque chaîne assure sa propre surveillance.

Remarque : La configuration double duale monitorée des servocommandes est rendue possible par les considérations suivantes :

* doublement des ordres de braquage de chaque chaîne CDVE
* utilisation des deux circuits hydrauliques par servocommandes
* contrôle possible de l'avion avec deux élevons, les autres étant braqués près du neutre (δm_0).

- La répartition des chaînes CDVE sur les gouvernes est présentée figure 5 ; elle se déduit des redondances présentées en préalable. Cette distribution permet de concilier sécurité et invulnérabilité, en créant un niveau de redondance complémentaire dû à la multiplicité des gouvernes utilisées en tangage et en roulis (4 élevons).

  . ELEVONS EXTERNES (et direction normale) contrôlés par les chaînes 2 et 3, contenues dans un seul Rack.

  . ELEVONS INTERNES (et direction secours) contrôlés par les chaînes 1 et 4, contenues dans un second Rack.

  . L'ensemble des gouvernes pilotées en modes secours (braquage d'équilibre, ou pilotage de toutes les gouvernes).

Ainsi, nous pouvons obtenir les modes de pilotage suivants :

* Pilotage bouclé de toutes les gouvernes en l'absence de panne ;
* Pilotage bouclé de toutes les gouvernes en présence d'une panne de chaîne ; il y a perte d'une redondance sur 3 gouvernes au maximum ;
* Pilotage de deux élevons (et éventuellement, direction) en mode bouclé, et positionnement de deux élevons à un braquage déterminé.
* Pilotage de toutes les gouvernes en mode secours, par la commande électrique de secours.

Remarque :

ON NOTERA QUE L'ENSEMBLE DES GOUVERNES DEMEURE CONTROLABLE APRES DESTRUCTION OU DEBRANCHEMENT DE L'UN QUELCONQUE DES CALCULATEURS DE CHAINES CDVE.

- Les gains automatiques sont élaborés dans un calculateur unique, alimenté par le réseau avion (E_3). Cette disposition permet de limiter les risques de destructions qui résultent des foudroiements très sévères ($dI/dt > 10^7$ A/s), en raison des liaisons que possède le calculateur avec les sondes anémométriques.

Les gains automatiques ont une grande importance pour pondérer les ordres de braquages des gouvernes, suivant le point de vol ; leur élaboration doit être sûre. En outre, les pondérations rendues possibles par le calcul fin des gains automatiques sont essentielles pour élaborer les fonctions de limitations automatiques retenues pour délivrer le pilote de tâches absorbantes et délicates. Ces limitations existent pour assigner l'avion à une incidence limite et à un facteur de charge maximal.

Nous avons donc retenu pour les calculs des gains fonctions du Mach et de l'altitude, trois processeurs numériques spécialisés, de notre conception, qui utilisent en synchronisme un logiciel unique. La validation complète de ce logiciel a été indispensable pour démontrer que nos objectifs de sécurité sont atteints. Cette structure triple processeurs, associée à un dispositif spécifique de gestion et de réarmement des pannes nous permet :

. de poursuivre le calcul des gains après panne simple ;
. d'arrêter le calcul des gains à la seconde panne,
. de réarmer automatiquement les premières et secondes pannes après leurs disparitions. Ce processus est fondamental en combat.

Enfin, le pilote dispose d'un sélecteur de gains programmés dans chaque chaîne CDVE si le calculateur est défaillant.

- La limitation automatique de l'incidence est très élaborée sur MIRAGE 2000. Son efficacité a été démontrée lors des essais en vol menés jusqu'à vitesse nulle. Cette limitation est réalisée dans les chaînes CDVE en fonction d'une mesure consolidée de la valeur d'incidence réalisée dans le calculateur des gains automatiques.

Elle résulte :

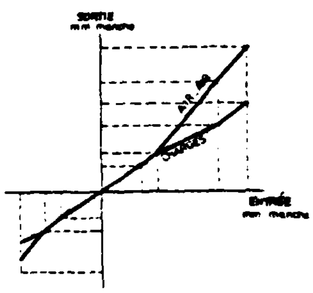
. de la mesure d'incidence droite,
. de la mesure d'incidence gauche,
. de la mesure d'une incidence-pression.

Pour des raisons analogues à celles exposées pour les gains automatiques nous avons isolé ces mesures susceptibles de foudroiements directs ; de plus nous avons retenu les mêmes dispositions de reconfigurations du système sur pannes :

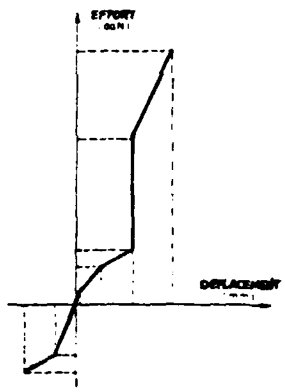
. réarmements automatiques des premières et deuxièmes pannes,
. commutation du limiteur automatique sur un dispositif de secours, dénommé "CORRECTEUR", qui est basé sur la mesure du facteur de charge dans les 4 chaînes.

- La limitation automatique du facteur de charge est assurée par la conjonction de trois facteurs et confirmée par les essais statiques structuraux :

. Utilisation des gains automatiques,
. Utilisation de l'information de présence des charges externes dans le schéma fonctionnel (schéma),
. Utilisation de lois d'efforts dissuasives au-delà de certains seuils mais qui assurent des manoeuvres de détresse (schéma).



Amedée Profondeur



Loi Effort-Deplacement
Manche en Profondeur

- Les signalisations au pilote présentent un caractère global, que nous avons défini pour supprimer tout risque d'interprétation sur l'état réel des dégradations du système. Cette disposition nous paraît fondamentale pour accroître la sécurité car :

. la gestion du système de signalisation est simple, donc a priori fiable
. les risques de fausse interprétation sont réduits,
. l'analyse des défaillances que nous avons effectuée permet de justifier les risques de dégradations successifs avec précision.

Les 8 voyants de pannes directement liés aux CDVE sont regroupés en banquette droite, et répétés par un voyant général de pannes situé très visiblement devant le pilote. Les voyants Rouge (qui indiquent une consigne impérative) sont doublés d'une alarme sonore. Le pilote dispose d'une commande de réarmement des sécurités qui peut dans certains cas réactiver toutes les voies du système.

- Les tests automatiques embarqués sont élaborés dans le calculateur numérique branché sur le réseau de l'avion. Il permet d'exécuter des séquences de contrôles internes des équipements qui sont très profondes.

Deux sortes de tests sont réalisables :

. le test automatique des sécurités et des calculateurs électroniques, réalisable sans énergie extérieure, comporte environ 2000 points de contrôles. Il est exécutable pour vérifier le système a priori, et pour les recherches de pannes mémorisées en vol ou au sol, ainsi que pour les dépannages.

. le test automatique de l'intégrité générale du système et des servocommandes est réalisable par le pilote en moins de 10 secondes, suivant un processus entièrement automatique.

COMMANDE DE SECOURS

- L'intérêt, ou la raison d'être d'une commande de secours à des CDVE redondantes est justifié par deux considérations essentielles :

. en combat, par les risques de destructions, d'incendies qui résultent d'impacts d'armes, d'explosion nucléaire à haute altitude ... etc ;

. en période de paix, par les risques de pannes multiples du système normal, ou par la défaillance d'un POINT COMMUN.

- La définition de la commande de secours a été retenue la plus simple, pour obtenir un niveau de fiabilité élevé et une maintenance simplifiée.

. La commande de secours est électriquement en attente, mais n'est pas active au niveau des servocommandes,

. après enclenchement, automatique ou manuel, le pilote dispose d'une commande de vol classique, de grande qualité, mais sans ordres de stabilisation artificielle ;

. les éléments de détection, de calcul et d'élaboration des ordres de la commande de secours sont séparés de leurs homologues des chaînes CDVE normales ;

. la commande de secours est traitée pour supporter les effets de la foudre et de l'IEM.

- L'enclenchement de la commande de secours peut être obtenu de deux façons :

. automatiquement.
. Manuellement, par le pilote.

L'enclenchement automatique peut entraîner deux modes distincts de fonctionnements, suivant *l'état de dégradation du système :*

. mode δ_{m_o} , utilisé sur deux élevons en présence de deux pannes à leur niveau
. mode secours rencontré isolément sur le gouvernail de direction après deux pannes à son niveau, ou sur les cinq gouvernes principales.

Ces enclenchements ont pour effets :

. de commuter les ordres électriques de commande au niveau des servo-moteurs électro-mécaniques externes de chaque servocommandes.

. de dépressuriser les servo-moteurs électro-hydrauliques normaux et secours de toutes les servocommandes.

JUSTIFICATION DU SYSTEME

Les justifications des CDVE du MIRAGE 2000 sont nombreuses et complexes. Elles sont groupées dans deux domaines complémentaires :

. les justifications pratiques
. les justifications théoriques.

- Les justifications pratiques ont été entreprises dès l'existence des matériels prototypes et ont concerné :

. les équipements ou composants
. le système complet
. l'avion complet

* Les justifications des équipements ou des composants ont été faites en laboratoires pour éprouver les matériels dans les conditions extrêmes d'emploi.

On notera en particulier les essais des composants électroniques à des chocs de courants représentatifs de l'IEM subie lors d'une explosion nucléaire à haute altitude.

* Les justifications du système complet ont été les plus importantes et les plus nombreuses :

  . Utilisation d'un système CDVE complet, lié à un simulateur d'étude, pour la mise au point, l'évaluation des pannes et des consignes.

  . Essai à la foudre d'un système CDVE représentatif dans une cellule complète d'avion :

    * En chocs de courant
    * En chocs de tension

  . Essai à l'IEM d'un système CDVE identique à celui évalué à la foudre, dans une cellule complète.

* Les justifications sur avion ont permis :

  . l'évaluation des systèmes en ambiance perturbée dans une chambre anéchoïde où l'avion est simulé en vol.

  . l'évaluation des pannes des CDVE et des générations associées, l'avion étant au sol, pour complément d'évaluation sur le simulateur d'études,

  . l'évaluation des pannes des CDVE en vol, jusqu'à l'enclenchement automatique (ou manuel) des modes dégradés et secours.

- Les justifications théoriques font l'objet d'une étude exhaustive des pannes qui a duré deux ans environ, et entraîné l'adoption de modifications.

Nous avons étudié le système sous différents aspects :

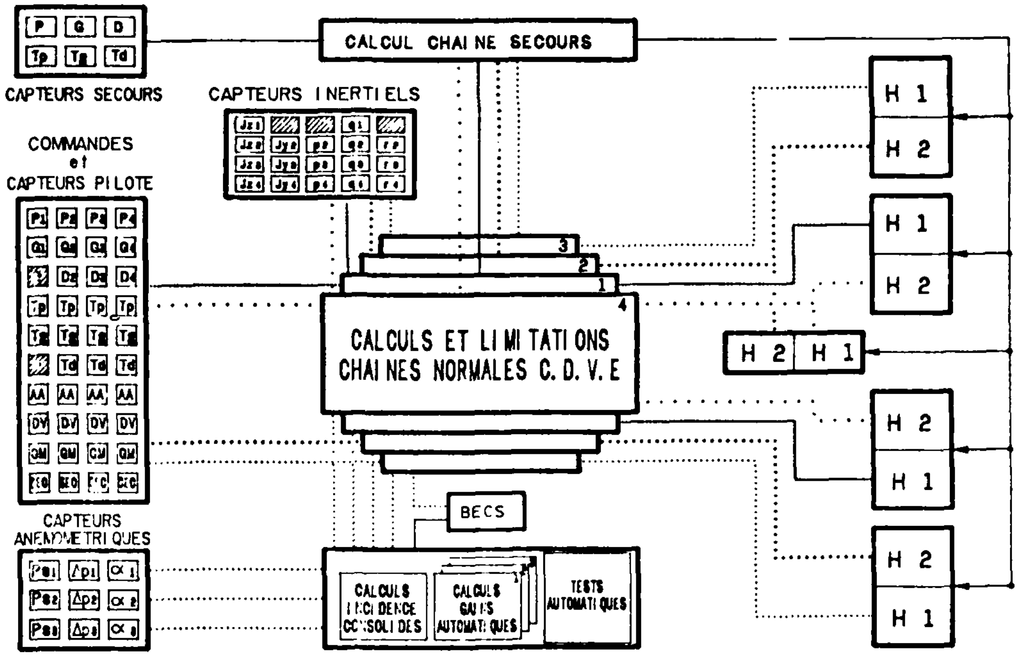
  . L'apparition et les effets des pannes successives pour une fonction, un axe de pilotage dans une chaîne, une chaîne de CDVE.

  . Le dénombrement et les justifications des points communs électriques, électroniques, mécaniques... etc

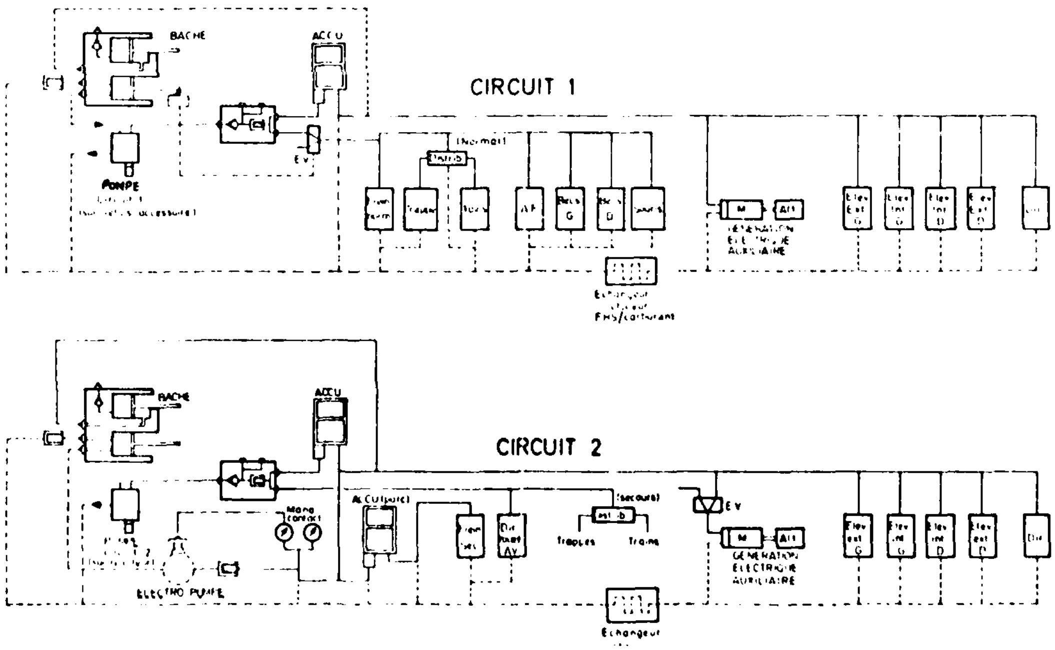
  . L'efficacité et la sûreté de la signalisation des pannes dédiée au pilote ou au personnel de maintenance.

  . l'efficacité et la sûreté des tests automatiques embarqués.

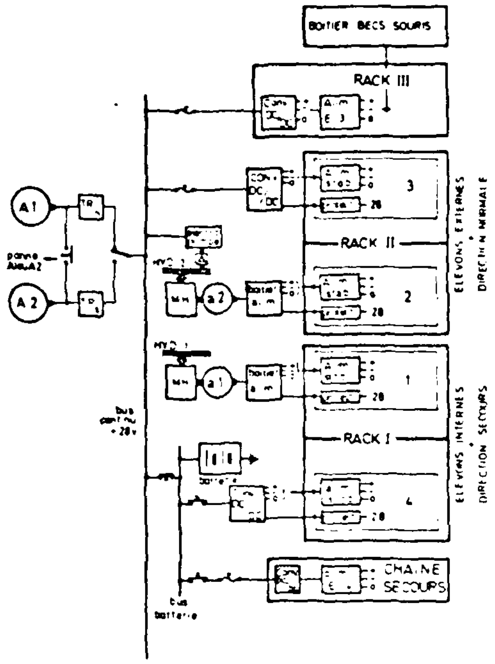
  . La détermination des opérations de maintenance et leur périodicité.



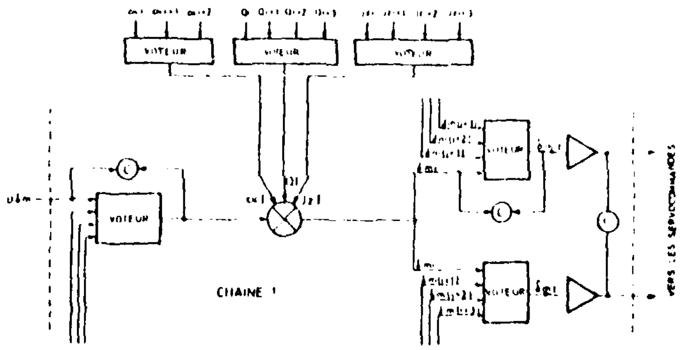
1. MIRAGE 2000 Redondance des C.D.V.E



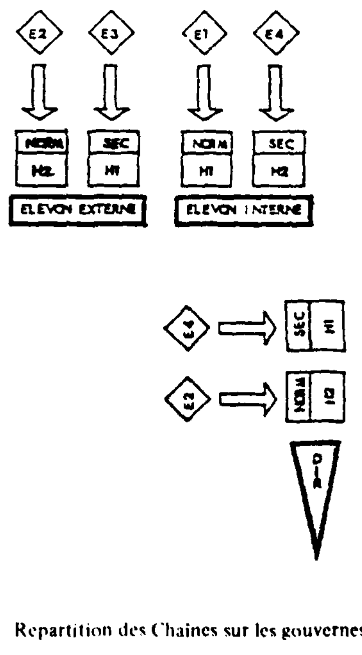
2. C.D.V.E Schema Hydraulique Avion



3. Repartition des Alimentations Electriques



4. Tronçonnement des voies C.D.V.E



5. Repartition des Chaines sur les gouvernes

TORNADO AUTOPILOT
MEASURES TO ENSURE SURVIVABILITY AFTER FAILURES

by
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SUMMARY

Measures applied to the autopilot of the TORNADO to ensure survivability of the aircraft after failures during automatic low level flying are presented. Apart from redundant equipment, these measures include hardware and software limiters to minimize the effect of failures upon the aircraft, hardware and software monitors to detect and isolate failures, emergency procedures to initiate recovery manoeuvres, as well as efficient testing of software.

1. INTRODUCTION

The Multi Role Combat Aircraft TORNADO is a modern military aircraft, which has recently gone into service with the RAF, GAF, GNY and IAF. One of the outstanding features of this aircraft is its capability to fly at low heights over land or sea, i.e. automatic terrain following/automatic radar height hold. The autopilot, as a central part of the flight guidance and control system, is responsible for performance, integrity and flight safety during automatic low level flying.

2. STRUCTURE OF THE TORNADO FLIGHT CONTROL SYSTEM

Fig. 1 shows a block schematic of the TORNADO Flight Control System.

The autopilot is a duplex redundant digital computer. The autopilot receives sensor signals from a number of sources: The primary control signals for the low level modes are supplied from the Terrain Following Radar (TFR) and the Radar Altimeter. The signals for guidance in the lateral plane are supplied from the Main Computer and the Horizontal Situation Indicator as well as the Inertial Navigation System (INS) and the Secondary Attitude and Heading Reference System (SAHRS). The INS and SAHRS are further used for attitude stabilization. Signals from the Air Data Computer (ADC) are used for scheduling purposes in the low level modes and for autothrottle control.

The autopilot receives further signals via the Command and Stability Augmentation System (CSAS): Rate gyro signals are used for turn coordination and improvement of dynamic response. Pressure signals from the Triplex Transducer Unit (TTU) are used for scheduling and monitoring purposes.

It should be noted that the signals from the Terrain Following Radar, Main Computer, INS, SAHRS and ADC are serial-digital, while the signals from the other sensors are analog or synchro signals which are converted into digital.

The autopilot outputs to the CSAS are pitch and roll rate demands. These signals are D to A converted and triplicated in the autopilot, since the CSAS is a triplex analog system. Further outputs are throttle demand to the autothrottle actuator, azimuth and elevation commands to the Head-Up Display and failure indications to the Central Warning Panel.

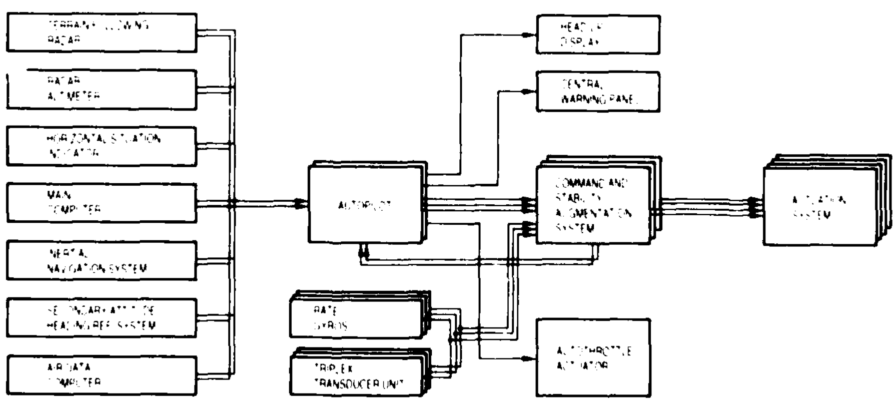

FIG. 1: BLOCK SCHEMATIC OF TORNADO FLIGHT CONTROL SYSTEM

3. SAFETY REQUIREMENTS

With regard to flight safety during automatic low level flying, it must be ensured that no single failure can lead to a hazardous situation, i.e. the aircraft under automatic control must be fail-safe. The probability of multiple failures must be less than a specified small value.

4. MONITORING FUNCTIONS AND AUTOMATIC EMERGENCY PROCEDURES IN THE BASIC DESIGN

The autopilot must be able to react to the following types of failures:

° Sensor failures which are self-detected by the affected sensors.
° Transmission faults of serial-digital data.
° Discrepancies between redundant sensors.
° Discrepancies between the commands in the two autopilot computers.

Sensor failures are indicated to the autopilot by a status bit or discrete signal. Transmission faults are detected by a parity check. Redundant sensor signals, i.e. attitude angles from INS and SAHRS and air data from ADC and TTU, are monitored against each other within the autopilot. The commands generated within the two autopilot computers are cross-monitored at various stages of the computation. Fig. 2 shows a block schematic of the cross-monitoring for the pitch command generation. The cross-monitoring for the roll axis is mechanized in a similar way.

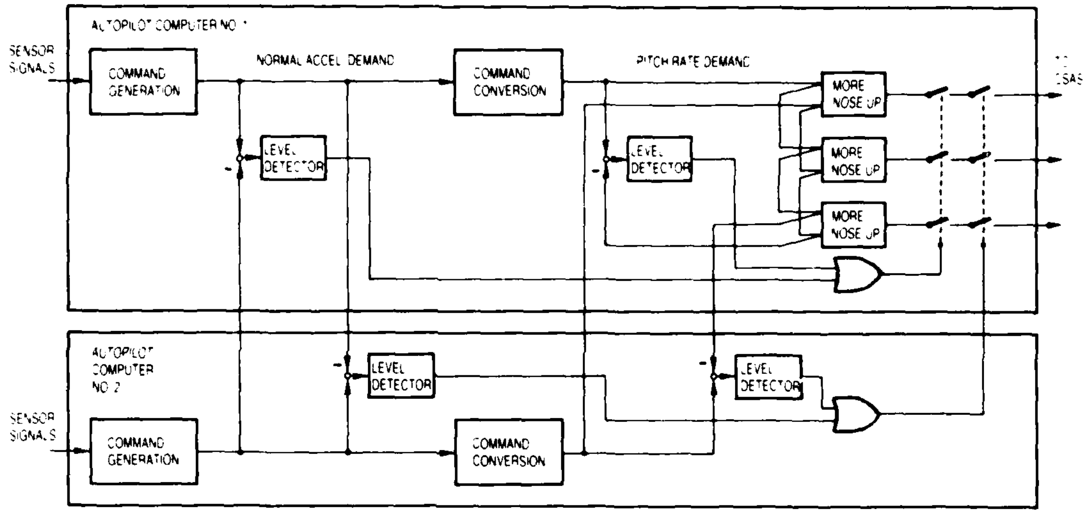

FIG. 2: BLOCK SCHEMATIC OF CROSS-MONITORING BETWEEN AUTOPILOT COMPUTERS

After detection of a failure the autopilot reacts as follows:
If one of the attitude sources indicates that it has failed, the autopilot automatically reverts to the healthy source and continues flying with unmonitored single source. The failure is indicated on the Central Warning Panel.

In all other failure cases the autopilot is automatically disconnected and a failure signal is transmitted to the Central Warning Panel, which initiates flashing of attention getters and audio warning. However, autopilot disconnect plus warning are not sufficient to recover the aircraft in all failure cases during low level flying. Therefore, immediately after disconnect of the normal autopilot commands, which might be affected by the failure, an automatic pull-up/wings level manoeuvre is initiated by discharging a store in each axis.

Fig. 3 shows how the autopilot disconnect and open loop pull-up/wings level command generation is mechanized: During normal operation the commands to the CSAS are tracking a command fader store in each axis. In addition, there is a pull-up store and a wings level store, which are tracked to a value equivalent to the available g-margin and the momentary bank angle, respectively. At autopilot disconnect, all four stores are isolated and discharged. The signals from the pull-up and wings level stores are superimposed to the signals from the command fader stores.

Fig. 4 shows the time histories of a typical pull-up/wings level manoeuvre.

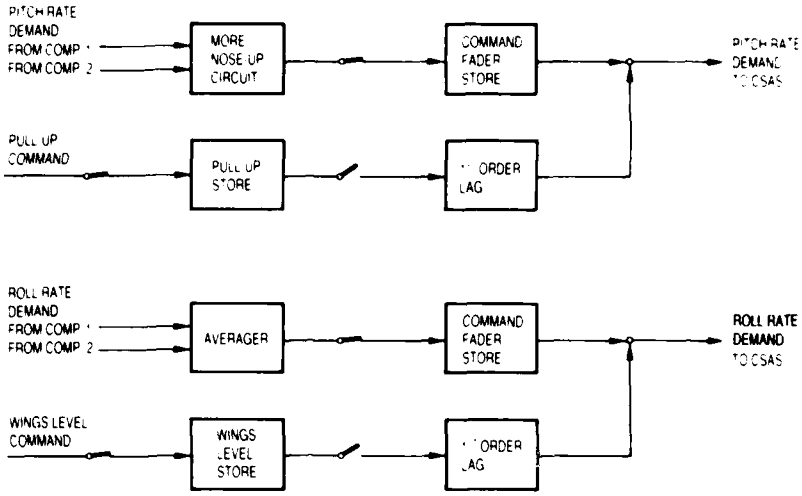

FIG. 3: AUTOPILOT OUTPUT SECTION

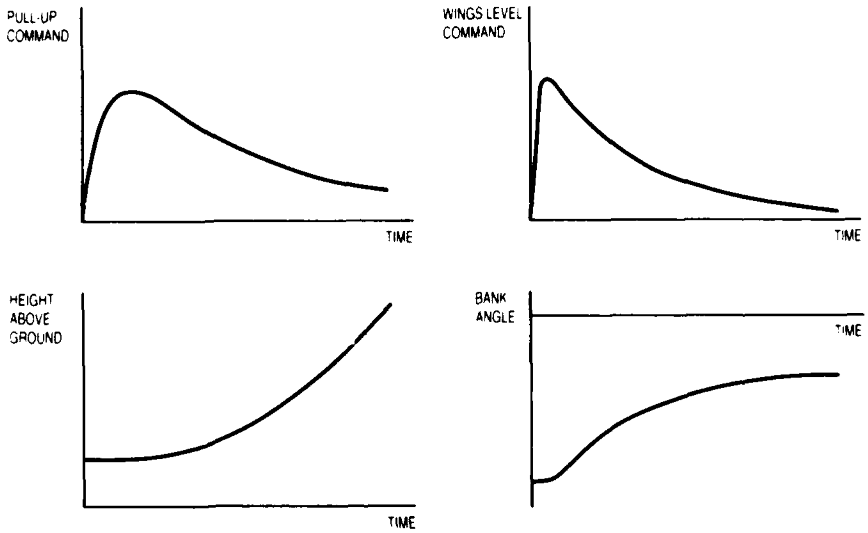

FIG. 4: OPEN LOOP PULL-UP/WINGS LEVEL PERFORMANCE

5. MEASURES TO IMPROVE SYSTEM INTEGRITY IN CASE OF COMPONENT OR SUBSYSTEM FAILURES

The pitch rate demand signal is in the final stage of the computation fed through an integrator. To avoid the integrator outputs of the two computers diverging, the crossfeed error is fed back into the integrators as shown in Fig. 5. A limiter on the crossfeed error ensures that the computer outputs are synchronized for small errors only. This synchronization is achieved at the expense of height offsets. To avoid negative height offsets, in particular in the low level modes, the symmetric crossfeed error limiter was replaced by an asymmetric one. Fig. 6 shows the resulting effect upon height offsets due to rate gyro signal offsets.

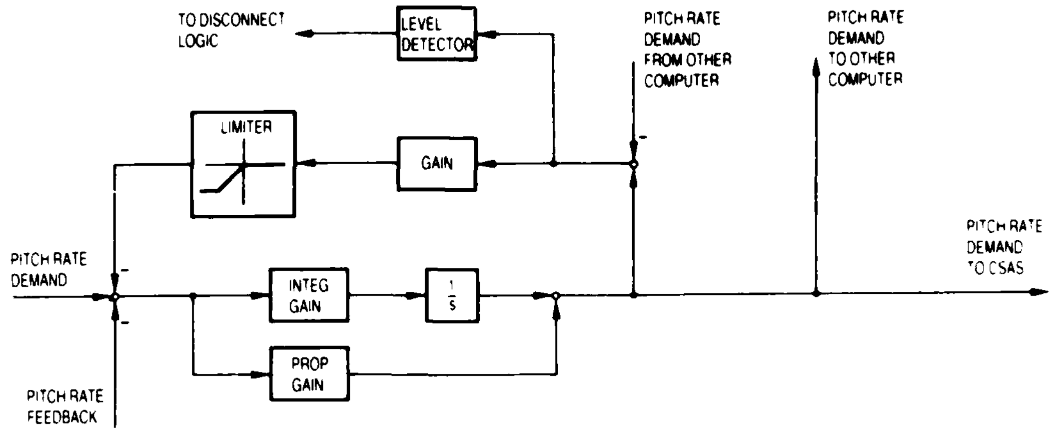

FIG. 5: SYNCHRONIZATION OF COMPUTER OUTPUTS

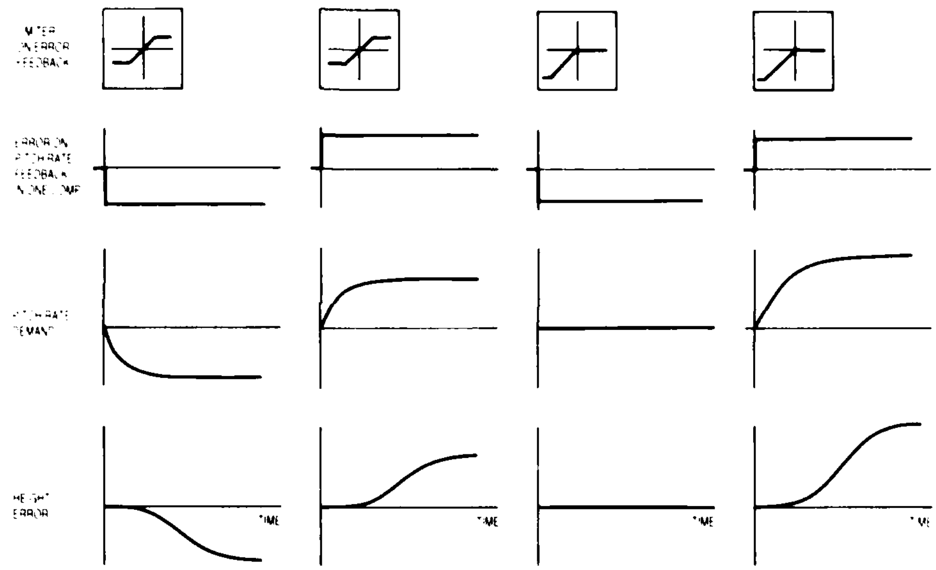

FIG. 6: HEIGHT OFFSETS DUE TO ERRORS ON PITCH RATE SIGNAL WITH SYMMETRIC AND ASYMMETRIC CROSSFEED ERROR LIMIT

The vertical acceleration demand signal in the autopilot is limited in accordance to the manoeuvring requirements for each mode. This limitation provides a protection against undetected sensor runaways. In the Terrain Following mode, these limits are relatively wide for both positive and negative demands. A negative runaway of the TF Radar command at low speed represents a potential hazard, as the "g" capability during a subsequent recovery manoeuvre is limited. Therefore, the negative vertical acceleration demand limit was scheduled with dynamic pressure to reduce the effect of potential negative runaways (Fig. 7).

In case of a detected TF Radar failure, automatic terrain following must be interrupted and a pull-up must be performed. In the basic design this was achieved by autopilot disconnect and subsequent open loop pull-up. However, since the autopilot is fully available in such a failure case, it was decided to introduce a closed loop pull-up/wings level facility, which provides a controlled recovery manoeuvre. Within 2 seconds the system is able to automatically return to TF, if the failure disappears. Otherwise, after 2 seconds the autopilot is disconnected. The performance of the closed loop pull-up is more consistent compared to open loop pull-up for different configurations and flight conditions. An example is given in Fig. 8.

Automatic recovery manoeuvres must not only clear the aircraft from the ground but also avoid unsafe flight conditions as a consequence of exceeding angle-of-attack limits. To ensure that the aircraft during automatic pull-ups does not build up excessive angles-of-attack, the normal acceleration demand limiter has to be scheduled with flight parameters and aircraft wing sweep (Fig. 9).

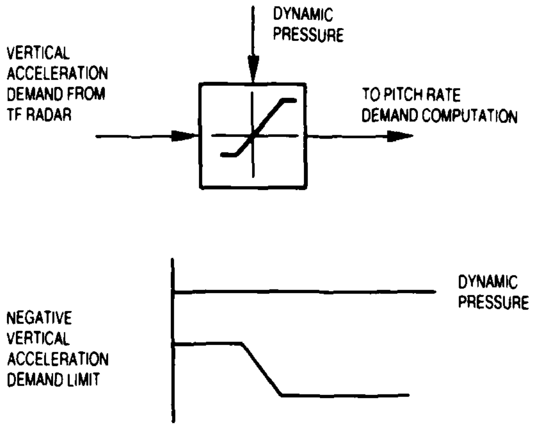

FIG. 7: NEGATIVE VERTICAL ACCELERATION DEMAND LIMIT SCHEDULING

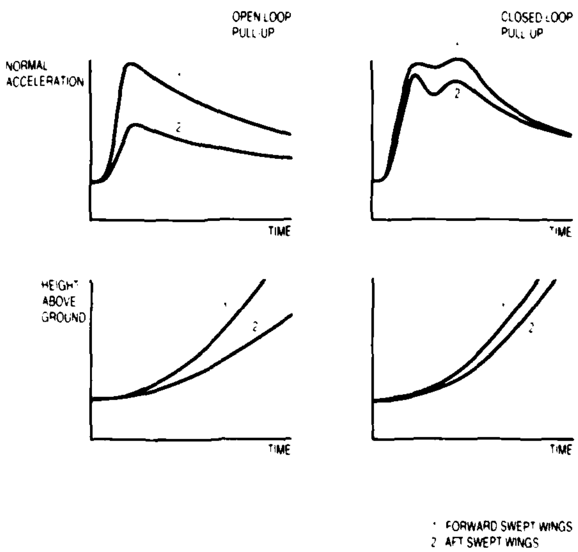

FIG. 8: PERFORMANCE OF OPEN AND CLOSED LOOP PULL-UP FOR TWO CONFIGURATIONS

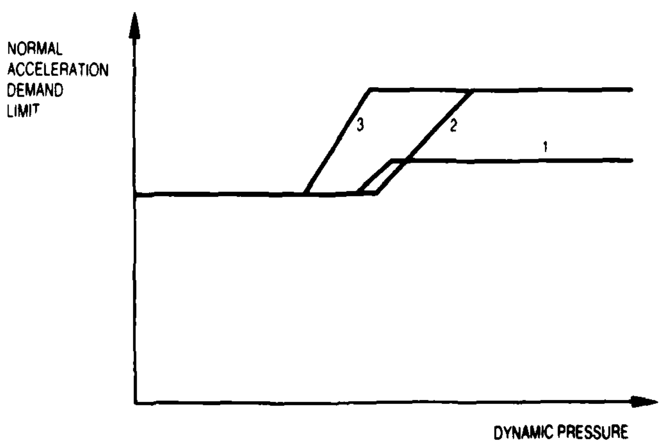

FIG. 9: NORMAL ACCELERATION DEMAND LIMIT FOR THREE DIFFERENT AIRCRAFT WING SWEEPS

An open loop pull-up during turning flight is always combined with an open loop wings level command, which is a function of the bank angle at the instant of pull-up/wings level initiation. In cases of a failure during manoeuvres with high roll rate, the wings level command with this type of control law is either too weak or too large to achieve a satisfactory levelling out of the aircraft. In order to improve the wings level performance under dynamic conditions, provisions have been made to include roll rate in the wings level command generation.

Signal discrepancies between the two computers are detected by the various crossfeed monitors, if the monitor thresholds are exceeded. This normally leads to an automatic autopilot disconnect. Since the detection and disconnect are controlled by software, a delay of up to one program cycle between exceedance of threshold and operation of output switches cannot be avoided. To prevent nuisance disconnects, the autopilot is disconnected only if a threshold is exceeded for a number of subsequent cycles (approximately 100 msecs). In case of a hardover in one computer, the commands will go right to the maximum value before disconnect. In the pitch axis, a positive hardover would, due to the more nose-up logic, appear on all three output channels and charge the fader stores. Such a hardover is not constrained by the normal acceleration limits and would lead to excessive transients of normal acceleration or angle-of-attack.

To protect the aircraft against hardovers, a hardover monitor has been introduced (Fig. 10). This monitor (one in each computer) is mechanized in hardware and reacts within nsecs. At detection of a hardover, the monitors open the output switches and thereby prevent the hardover being transmitted to the CSAS and stored in the fader stores. The permanent disconnect is carried out as before with a 100 msec's delay. Therefore single spikes are suppressed but do not lead to nuisance disconnects. Fig. 11 shows the size and duration of discrepancies which are tolerated by the monitoring system. Discrepancies exceeding the given boundaries lead to error suppression and/or disconnect.

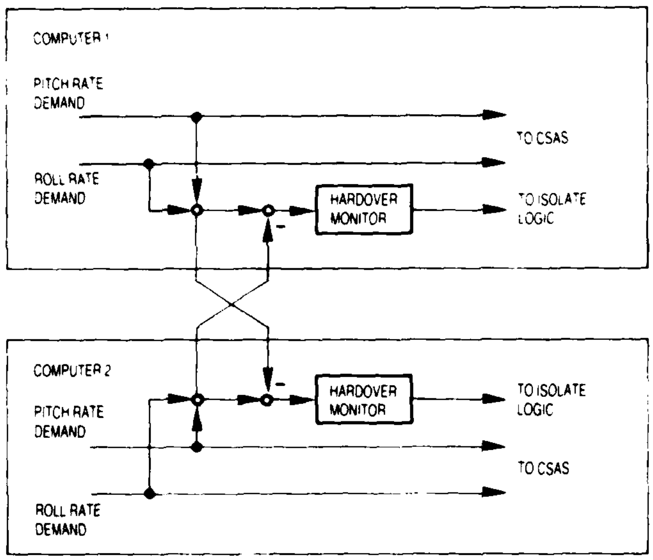

FIG. 10: BLOCK SCHEMATIC OF HARDOVER MONITOR

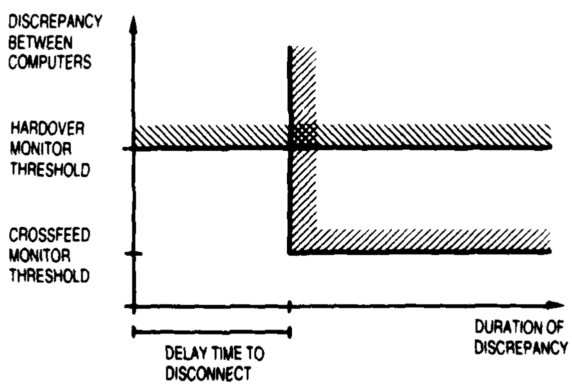

FIG. 11: AREA OF ERROR TOLERATION

The hardover monitor gives protection against common channel hardovers in the output lanes due to single failures of hardware components. However, it would not cover the case of a simultaneous hardover on all output channels due to a discrepancy in the autopilot software, which is common to both autopilot computers. A nose-down hardover of this kind could lead to excessive height loss and possibly a loss of the aircraft during low level flight.

To protect the aircraft and crew against such a remote possibility pending completion of all software testing, authority limits on the autopilot outputs were introduced and the disengagement characteristic was optimized. The general requirements for this design change were as follows:

° The pilot's authority should be as high as possible compared to the autopilot. The autopilot authority can only be reduced to a level which still ensures full performance during normal operation including safe recovery after any kind of failure.

° The task of the output faders is to reduce normal disengage transients. However, the faders also oppose a pilot's recovery stick input. Therefore, the fader time constants can only be reduced to a level which still ensures adequate suppression of transients after normal autopilot disengagement and at the same time gives the pilot immediate control over the aircraft if an emergency recovery action is required.

The authority limits were determined to allow for a 3 g incremental command during closed loop pull-up and a -1 g pushover command during normal TF.

It has been demonstrated by simulation that the authority limits and the reduced fader time constants would considerably reduce the height loss during recovery after a common channel nose-down hardover. Therefore, the introduction of this change was an important step to get the aircraft cleared in the automatic low level modes.

6. MEASURES INTRODUCED TO ENSURE SYSTEM INTEGRITY WITH REGARD TO THE SOFTWARE PROGRAM

The introduction of autopilot authority limits and minimization of output fader time constants was an important contribution to safety in the presence of multiple channel failures, but was never regarded as a replacement for rigorous testing of the software. The verification of error-free software was a prerequisite for getting the aircraft released down to minimum set clearance height.

To be able to detect any possible discrepancies within the software program, a so-called Cross-Software Test System was developed (Fig. 12). For this purpose, the autopilot control laws and mode and failure logic were once more programmed in a dissimilar manner, i.e. by using

- ° An independent programming team,
- ° A dissimilar programming language, i.e. High Order Language C with floating point arithmetic,
- ° A dissimilar computer, i.e. a PDP 11/70.

The test computer beside the autopilot model includes stimulation functions and evaluation routines to compare autopilot and model outputs against each other.

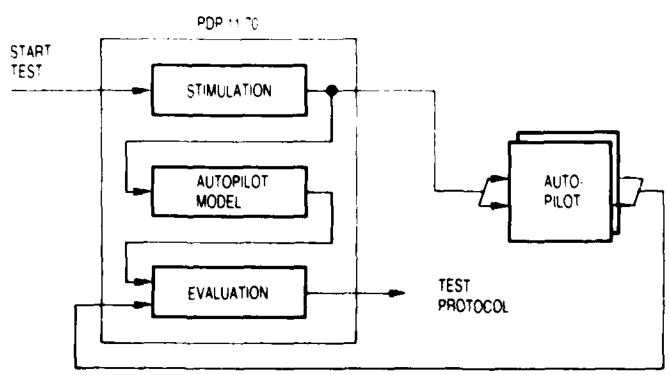

FIG. 12: BLOCK SCHEMATIC OF CROSS-SOFTWARE TEST SYSTEM

In particular, tests were carried out in the following areas:

- Variation of input parameters within full range,
- Initialization of integrators, filters and limiters,
- Mode switching.

A complete test included 50 hours of testing with a total of 5.6 million test points.

The efficiency of the cross-software test method was demonstrated by the fact that a number of minor software discrepancies were detected with this test, which fortunately had no significant effect upon safety and performance. These discrepancies had not been found during years of ground and flight testing. The confidence in the software gained by this test was the necessary final step to obtain full clearance of the low level modes.

7. CONCLUSION

The following measures have proven to be effective to ensure survivability of the aircraft after failures:

(1) As a basic requirement safety critical subsystems (autopilot computer, sensors) have to be made redundant to allow for cross-monitoring and detection of single component failures. After detection of a significant failure emergency procedures must be automatically initiated, i.e. automatic autopilot disconnect, automatic pull-up and wings level manoeuvre and adequate warnings to the pilot.

(2) The effect of failures must be suppressed or reduced by command limits in the software, hardover protection and output authority limitation. Automatic recovery manoeuvres must be optimized to give the pilot optimum conditions for take-over.

(3) Software integrity must be verified by efficient testing.

Practical flying experience has shown that the above safety measures are efficient to ensure flight safety during automatic flying including low level modes.

CERTIFICATION EXPERIENCE OF THE JAGUAR FLY-BY-WIRE DEMONSTRATOR AIRCRAFT INTEGRATED FLIGHT CONTROL SYSTEM

by

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SUMMARY

This paper briefly describes the digital Integrated Flight Control System (IFCS) developed for the Jaguar Fly-By-Wire (FBW) demonstrator programme, identifying the specification requirements, resultant architecture, implementation and the incorporated self test capability. The redundancy management aspects of the IFCS are described together with the techniques for providing the pilot with relevant information to determine the IFCS redundancy status. Particular emphasis is given to the software definition and preparation procedures, and the comprehensive integrity appraisal leading to flight clearance of the system.

Following the extensive rig proving of the system, the early phases of flight test were very successfully carried out using the fixed gain control laws. During this period a major software update was commenced to incorporate the scheduled gain control laws and to enhance the self test capability. The software segregation introduced at this stage is described, together with the experience obtained in recertifying the system. Flight testing of the scheduled control laws is continuing, and the minor problems encountered are mentioned. A further software revision to include the control laws for the statically unstable aircraft is well advanced, and the benefits of software segregation identified during this revision are described.

The reliability of the aircraft and IFCS have proved, to date, to be excellent. Thus practical in-flight results of the systems ability to absorb and survive fault conditions are minimal. The redundancy management and integrity experience provided by the programme has therefore principally been in the theoretical analysis supported by controlled experimentation on the rig. These exercises have highlighted key areas of the system and software design techniques which enable these aspects to be fully and economically evaluated. These areas are described, with mention of how these techniques are being developed to simplify and improve the exercise for future high integrity digital flight control systems.

1. AIRCRAFT AND SYSTEMS DESCRIPTION

1.1. Aircraft

The FBW Jaguar demonstrator aircraft is a modified single seat SEPECAT Jaguar. Internal modifications were made to accommodate the IFCS computers and extensive instrumentation, and all of the original mechanical control rods, autostabiliser equipment and powered flight control units were removed. A third Transformer Rectifier Unit (TRU) was added to cover the additional loading of the fly-by-wire system and instrumentation, and revised power distribution was introduced to meet the power supply integrity requirements of the IFCS.

Three independent 28V bus bars, each battery backed, are supplied by the three TRU, and each computer of the IFCS consolidates power from two of these bus bars as shown in Figure 1.

The two engine driven hydraulic pumps were replaced by units with greater capacity, and the emergency electrohydraulic pump was replaced by two pumps of greater capacity each driven by one of the independent, battery backed, 28V bus bars. These provide two independent hydraulic systems, each with an emergency supply primarily to power the flying control actuators, the system including provision for priority valves if found necessary. The standard power transfer unit, allowing transfer of power but not fluid between systems, is retained.

Externally the aircraft is little changed, though later in the trials programme leading edge strakes will be fitted along the air intake boxes. Provision is made for fixed ballast to be carried in the rear fuselage, and this together with fuel management procedures enable the centre of gravity to be moved aft to give a manoeuvre point of $-3\%\bar{c}$ to $-5\%\bar{c}$. The leading edge strakes will move the centre of pressure forward to give a manoeuvre point of $-10\%\bar{c}$.

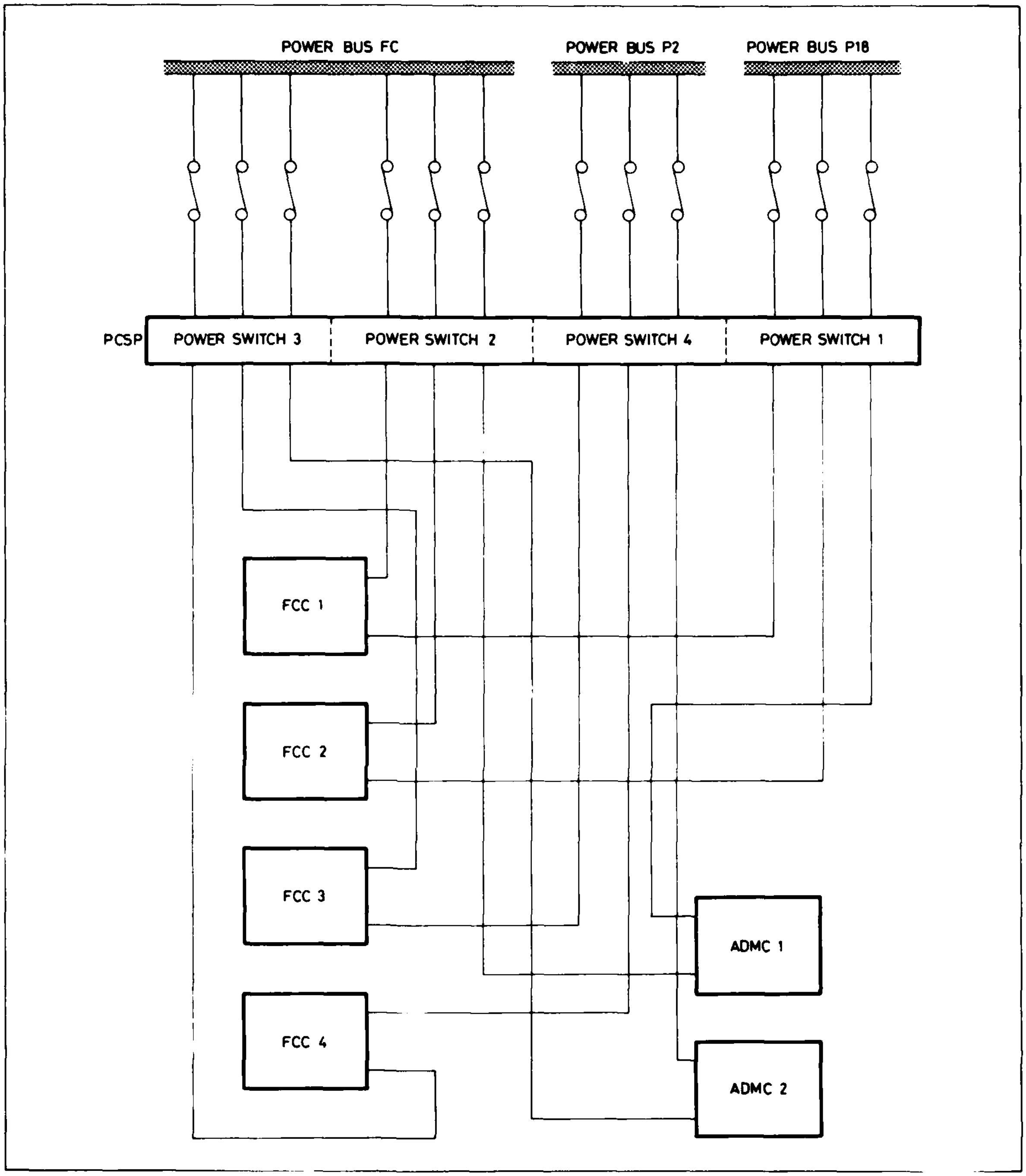

Figure 1 Flight Control System Primary Power Distribution

1.2. Integrated Flight Control System

The system architecture shown in Figure 2 was evolved to meet the following specification requirements.

- Overall system loss probability (including first stage actuation) shall be no greater than 10^{-7} per hour.
- Any two electrical failures in the system shall be survived.
- The electrohydraulic first stage actuation would have only two independent hydraulic supplies with no interconnection between them.
- The system shall survive a hydraulic system failure followed by an electrical system failure or an electrical failure followed by a hydraulic failure.
- The system shall rely on majority voting of the redundant elements for failure survival rather than self-monitoring within each redundant element.
- Similar redundant digital implementation shall be adopted without any reliance on any back-up flight controls (e.g. mechanical or analogue links).

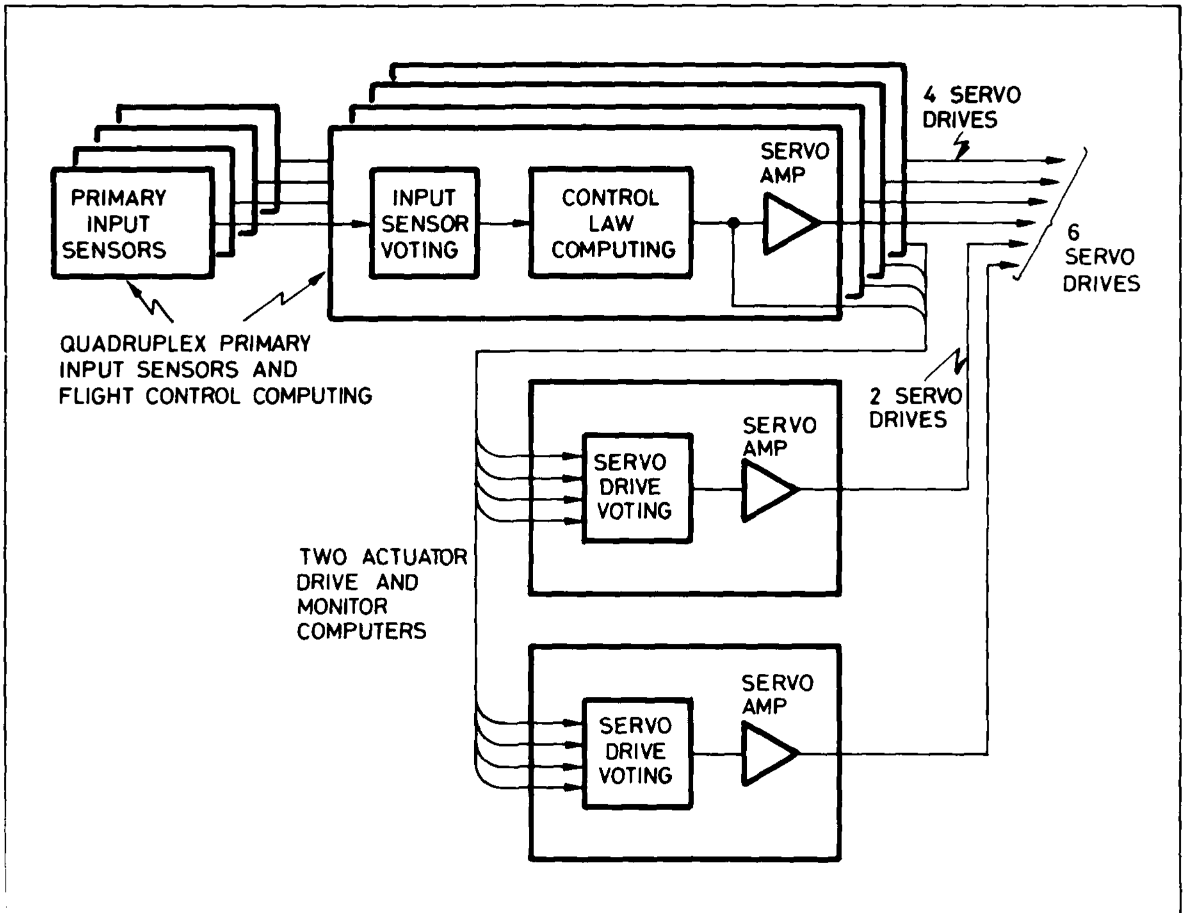

Figure 2 System Architecture

These requirements led to the incorporation of the duo-triplex actuation system, developed by Dowty Boulton Paul, to drive the rudder, two taileron and two spoiler control surfaces. The five Powered Flying Control Units (PFCU) are essentially similar, with variations in valve ports, jack strokes and diameters. Each PFCU, schematically shown in Figure 3, contains six flapper nozzle servo valves which convert electrical inputs from the Flight Control Computers (FCC) and Actuator Drive and Monitor Computers (ADMC) into hydraulic signals which are then used to drive a pair of first stage spool/control valves. Each servo valve is connected to two pairs of opposing pistons inside one of these first stage actuators. The pistons act on flanges mounted on the actuator spools, two pairs being used to prevent assymmetric loading. Both flanges are therefore driven by six pairs of opposing pistons, two pairs from each of three servo valves. A mechanical link between the two actuators ensures that the spools and thus the flanges move in unison, so that all six servo valve outputs are effectively summed together. In this way failures in at least two lanes can be absorbed within the actuators, the four good lanes overriding the failed lanes.

A separate hydraulic supply feeds each trio of servo valves associated with the first stage actuators and is also routed through the first stage actuator to the corresponding jack of the conventional tandem power control unit. Thus failure of either hydraulic supply can be tolerated by the PFCUs in addition to at least one electrical failure that affects the computing driving the side of the actuator unaffected by the hydraulic fault.

This actuation architecture requires 6 independent drive signals to each actuator, but the remaining integrity objectives do not necessitate the cost and complexity of a full six lane system. The Flight Control System (FCS) is therefore essentially a quadruplex digital system with special facilities to provide the additional independent drives to the actuators. All mechanical rods downstream of the trim and feel units have been removed, thus there is no mechanical or independent back up reversion.

Quadruplex Position Sensors (QPS) are used to sense pilot control demands in terms of stick, pedal and trim inputs and quadruplex rate gyros sense aircraft pitch, roll and yaw rates. Four identical digital FCC are used to process these signals together with those from other sensors. The resulting command signals are used to control the actuators. To convert the quadruplex signals from the FCCs into the sextuplex signals required by the actuators, the FCCs are supplemented by two ADMC.

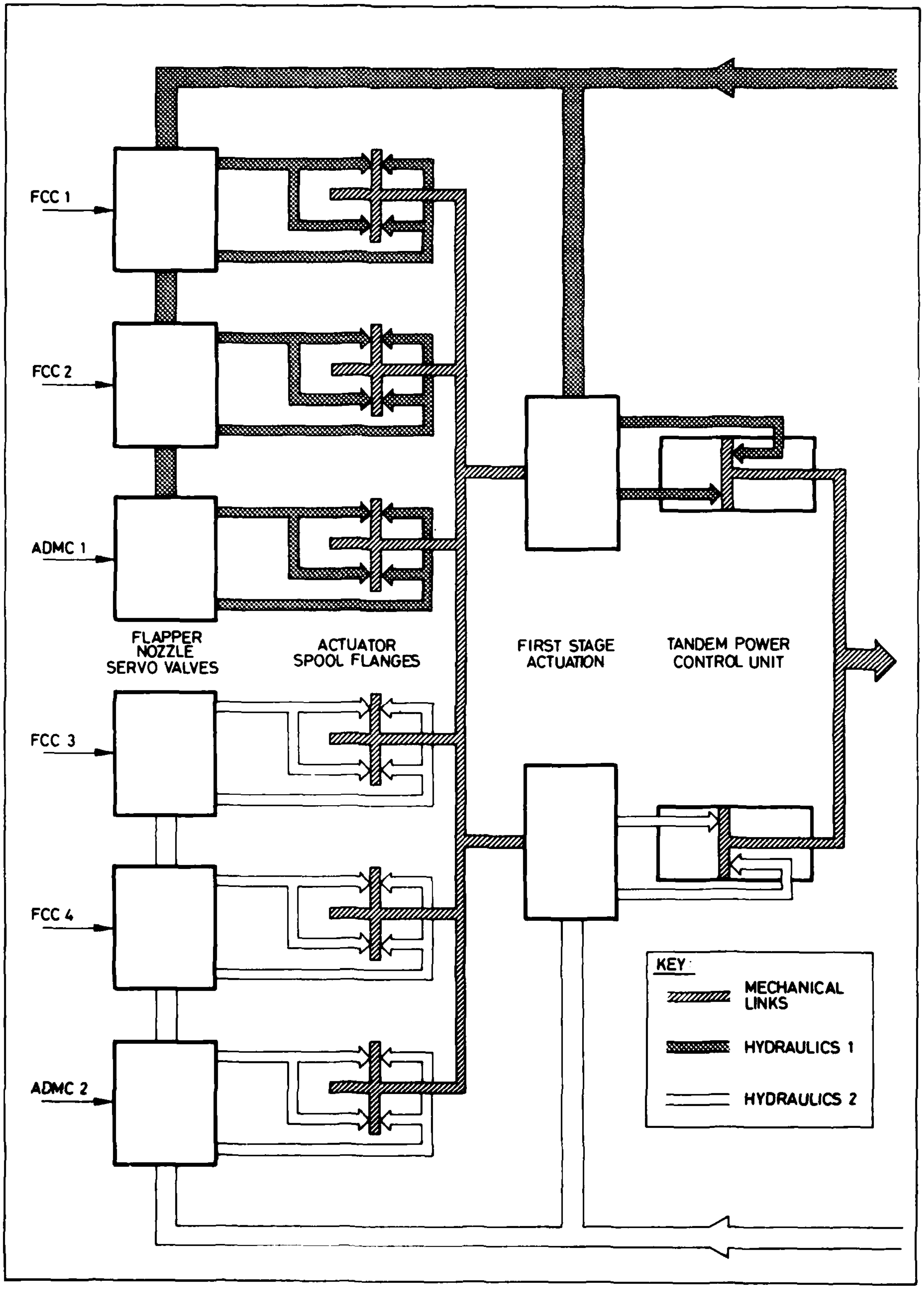

Figure 3 Duo-triplex Actuator Scheme

Figure 4 shows the schematic of an ADMC which receives optically coupled signals from all four FCC, converts them to analogue and votes them to provide a consolidated, essentially independent output.

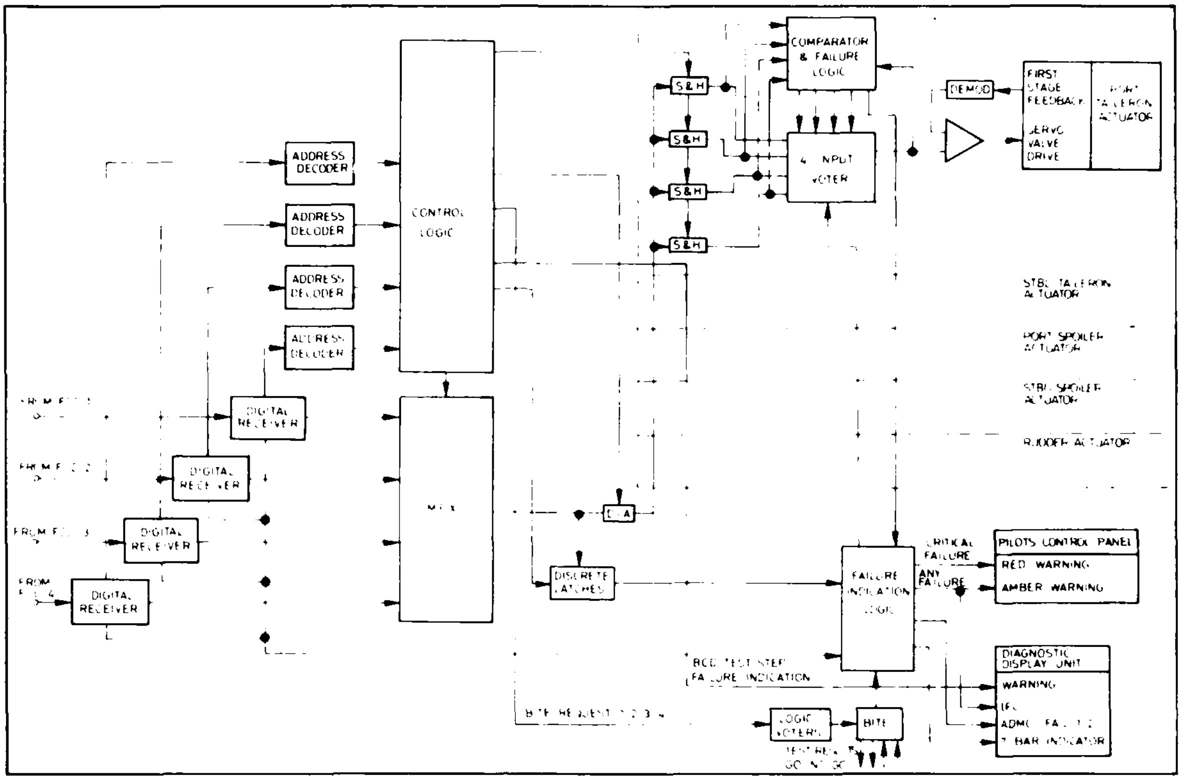

Figure 4 Actuator Drive and Monitor Computer Schematic

The Jaguar FBW system configuration is illustrated in Figure 2 which presents a simplified schematic of the primary control path. In addition to the quadruplex primary input sensors, sensors of lower redundancy are used for those functions which may be necessary for optimum handling qualities but which are not necessary for safe flight. These are dynamic pressure, static pressure, incidence and sideslip which are all triplex sensors; and lateral acceleration, flap position and airbrake position which are duplex sensors. Triplex dynamic and static pressures are provided by three pitot static probes (the standard nose boom and two side mounted probes). Triplex incidence and sideslip signals are provided by four Airstream Direction Detector probes (ADD) mounted around the nose of the aircraft.

The FCS also uses a number of quadruplex and duplex discrete inputs for switching functions. A simplified overall system configuration is illustrated in Figure 5. Cross lane data transmission is achieved via dedicated, optically coupled serial data links as shown in Figure 6. Voting and failure rejection logic in each computer maximises the system failure absorption capability and ensures the the system is able to survive two sequential failures of all primary input and feedback sensors. The system is designed to run synchronously, but has been operated assynchronously for considerable periods without observable degradation of performance. A more detailed description of the system architecture and the system LRUs can be found in reference 1.

The system includes comprehensive Built-In-Test (BIT) features which were specified to provide an accurate, decisive, and repeatable method of measuring equipment functional characteristics. In particular the BIT is used to clear the system in the aircraft prior to each flight thus its integrity and fault detection ability have to be compatible with the overall integrity of the system. The facility developed has met the objectives and provides an invaluable aid to FCS commissioning on the aircraft and reclearance of the FCS following Line Replaceable Unit (LRU) changes. A pilots Control and Switch Panel, shown in Figure 8, provides system status indication to the pilot. Status signals from the Flight Control Computers are consolidated to illuminate a STATUS amber warning (first failure) or red warning (similar second failure). The pilot may attempt a reset, when an amber warning is given, by pressing the STATUS button. If the detected disparity is no longer present the system will return to full operation status and the warning is extinguished. A red warning inhibits the status reset facility. Separate status indicators are provided for the secondary sensors. The panel also carries the autopilot engage buttons, the BIT initiate button, a facility to select different control law gains, and power switches to isolate the supplies to the computers to enable pre-flight check of the power supply consolidation within these units.

The FCS Equipment has been developed to production standards, as shown in Figures 7, 8, 9 and 10, and qualification tests have been completed.

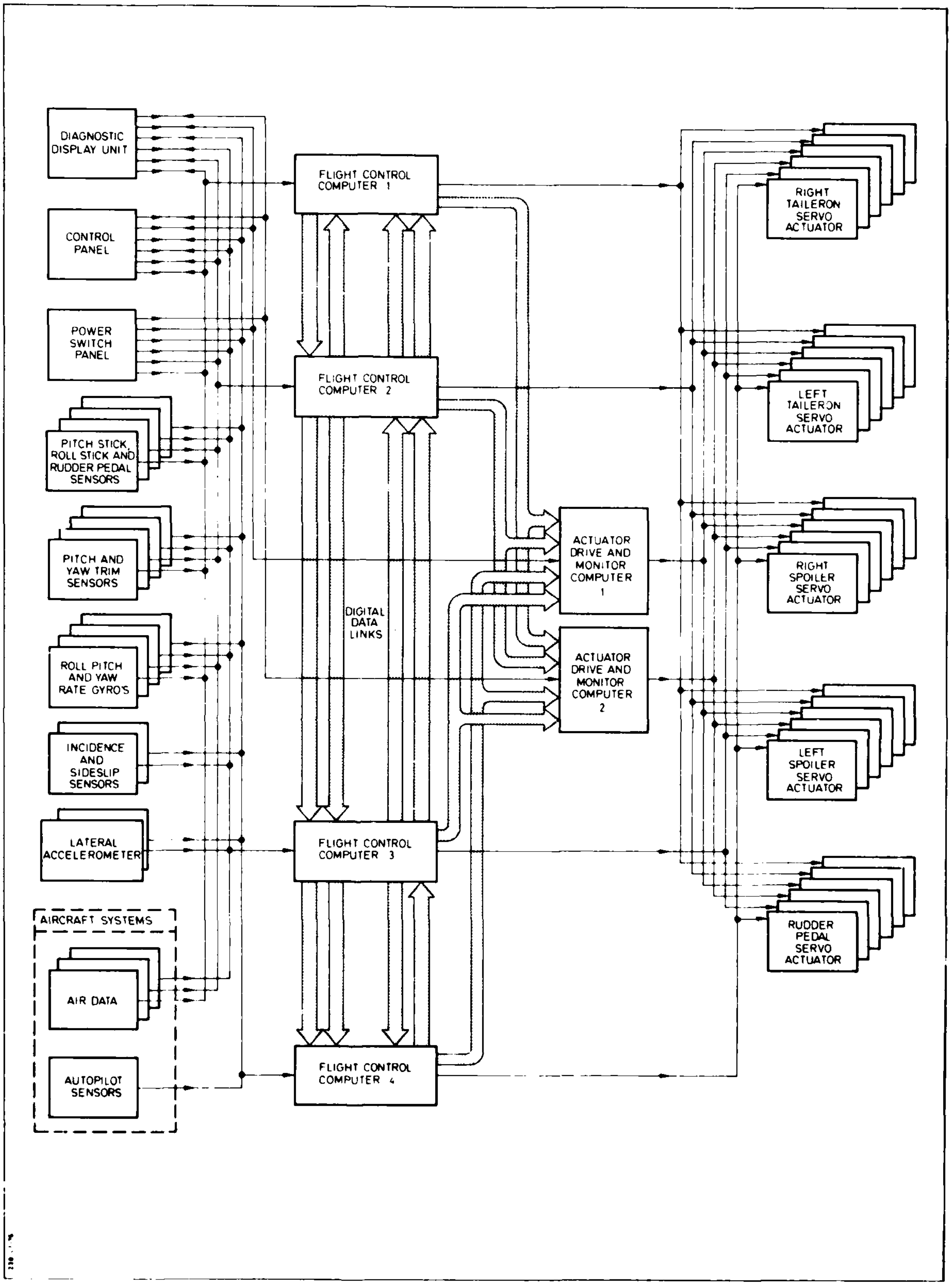

Figure 5 Flight Control System Configuration

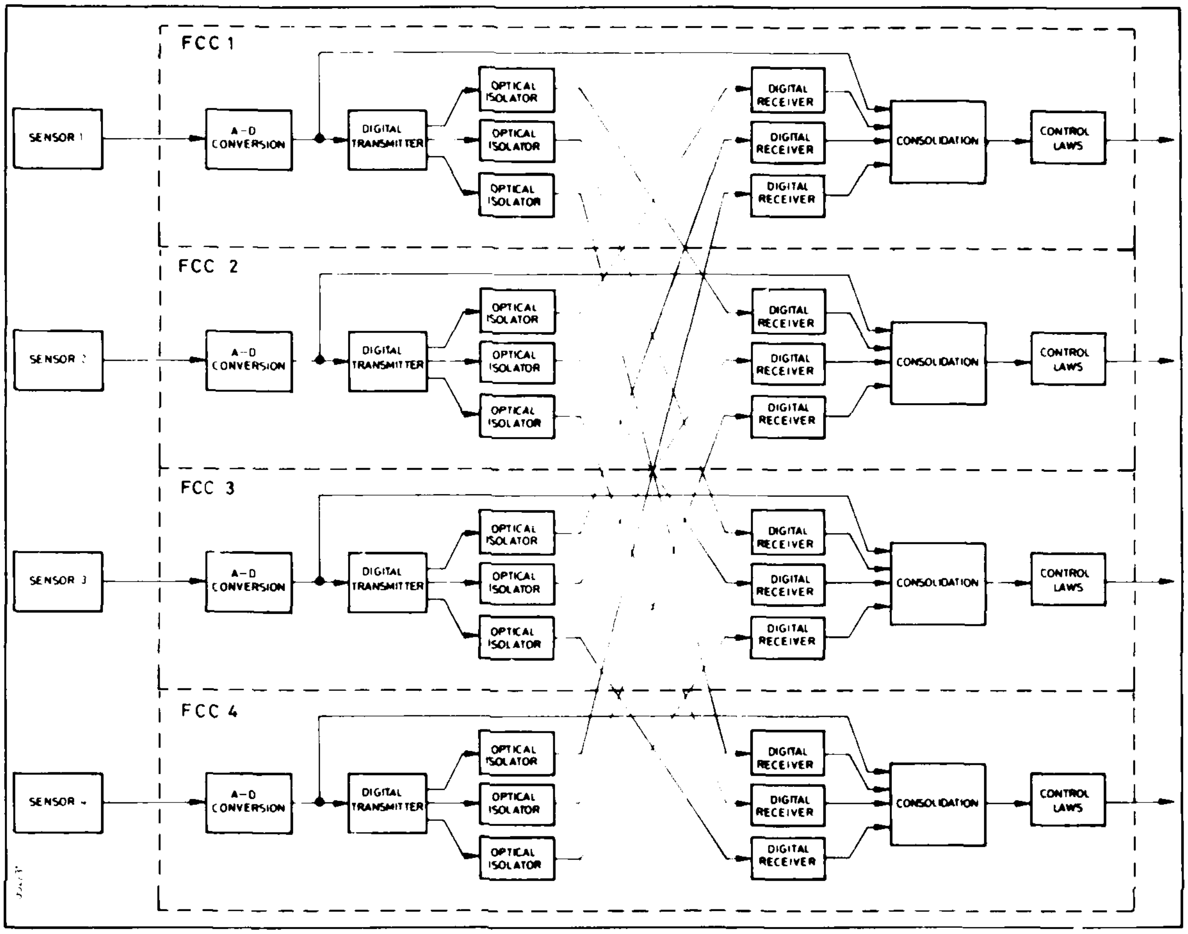

Figure 6 Optically Coupled Cross Lane Data Transmission

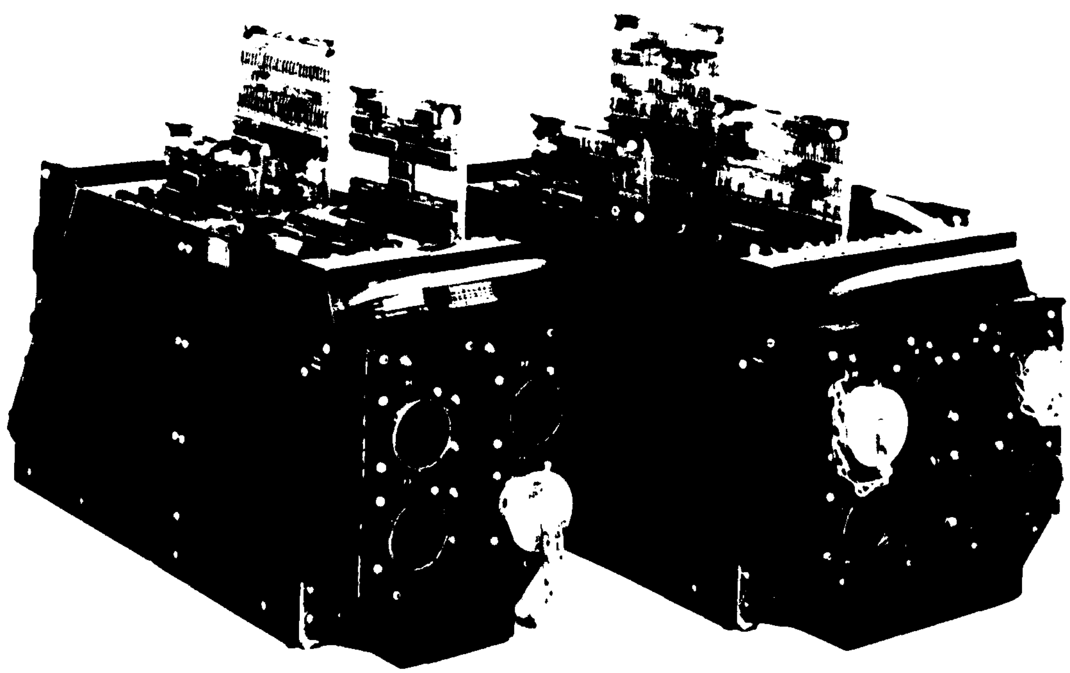

Figure 7 Actuator Drive and Monitor, and Flight Control Computers

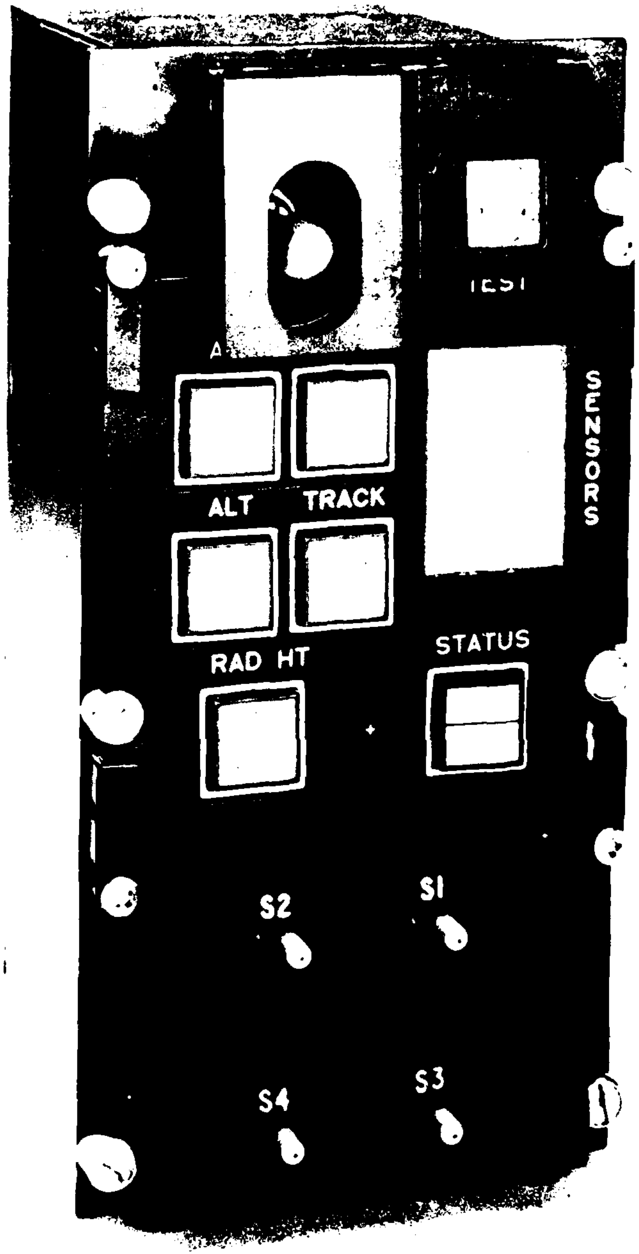

Figure 8 Pilots Control and Switch Panel

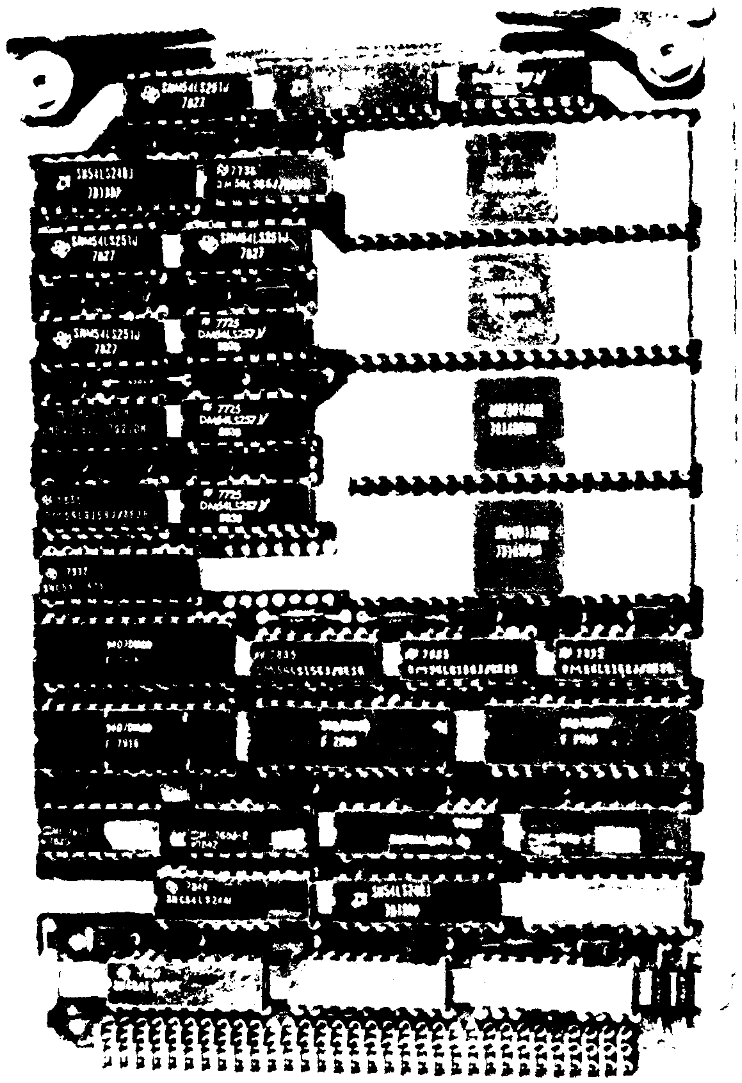

Figure 9 Typical Computing Module

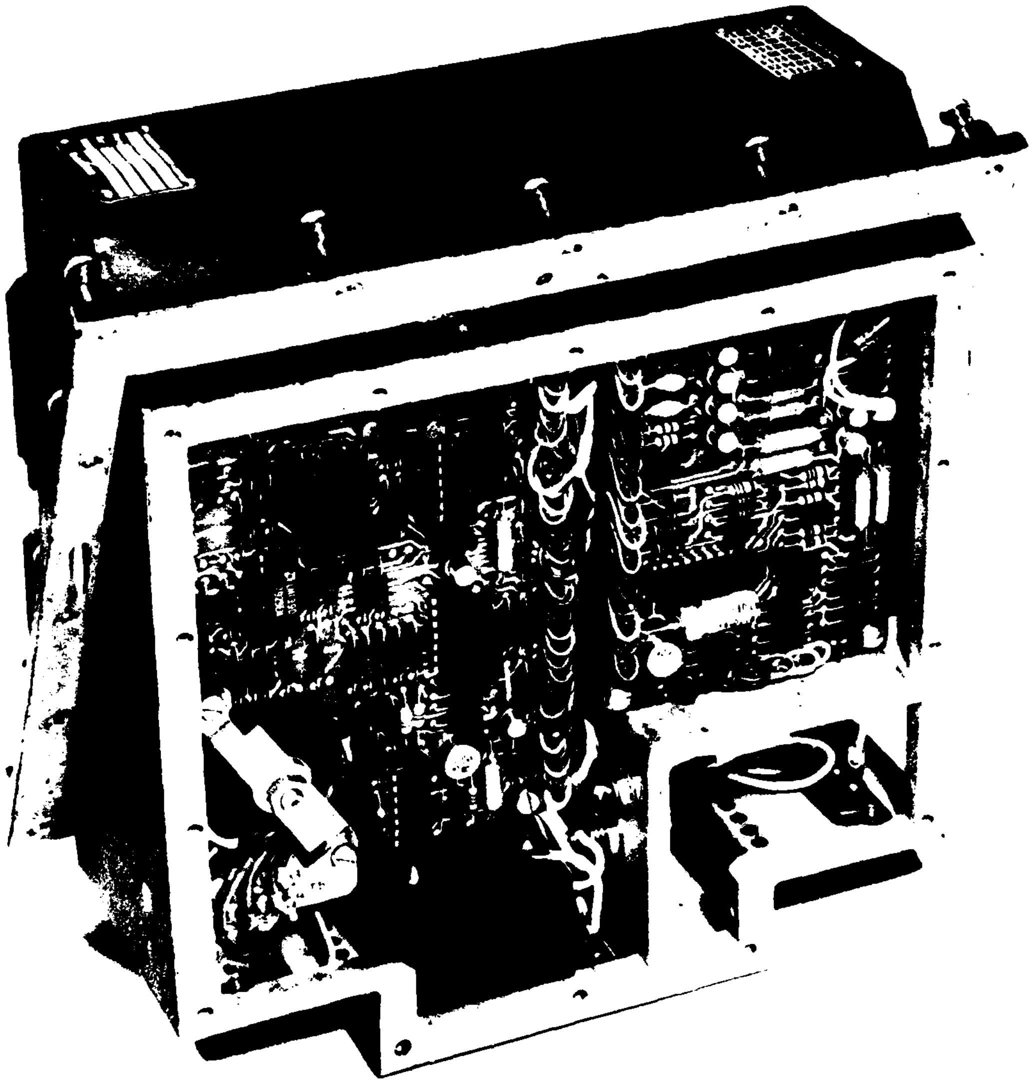

Figure 10 Power Supply Unit

2. FLIGHT RESIDENT SOFTWARE

2.1. Introduction

Both the system specification requirements, and cost/timescale considerations dictated the use of common high integrity soft-ware in all lanes of the FCS. There is, therefore the possibility of introducing design limitations via the software that could result in a common mode malfunction of the system and a subsequent safety critical loss of control. To contain this problem software structures and design procedures have been evolved over several digital FCS programmes. These maximise the visibility of the software to facilitate thorough test and functional audit during the design phase. These are supplemented by clear requirements definitions, detailed documentation, and rigorous production and configuration control procedures.

2.2. Flight Software Organisation

The real time control is achieved by a hardware Master Reset Timer which calls a non interruptable Executive. The Executive then calls the Frames (processing time slices containing related functional modules) in a defined sequence to provide the required iteration rates for the various computing paths. Each Frame typically contains control laws, with related signal selection and logic module functions, and consists of a set of program modules each defining a function that is easily defined, implemented, tested and audited. The worst case run time of a Frame is controlled at the design stage to ensure that the computing task is completed before the Master Reset occurs. Should any fault occur that causes the Frame run time to exceed the Master Reset time interval, this is detected and flagged as a computer fault.

The structure of the flight resident program is shown schematically in Figure 11.

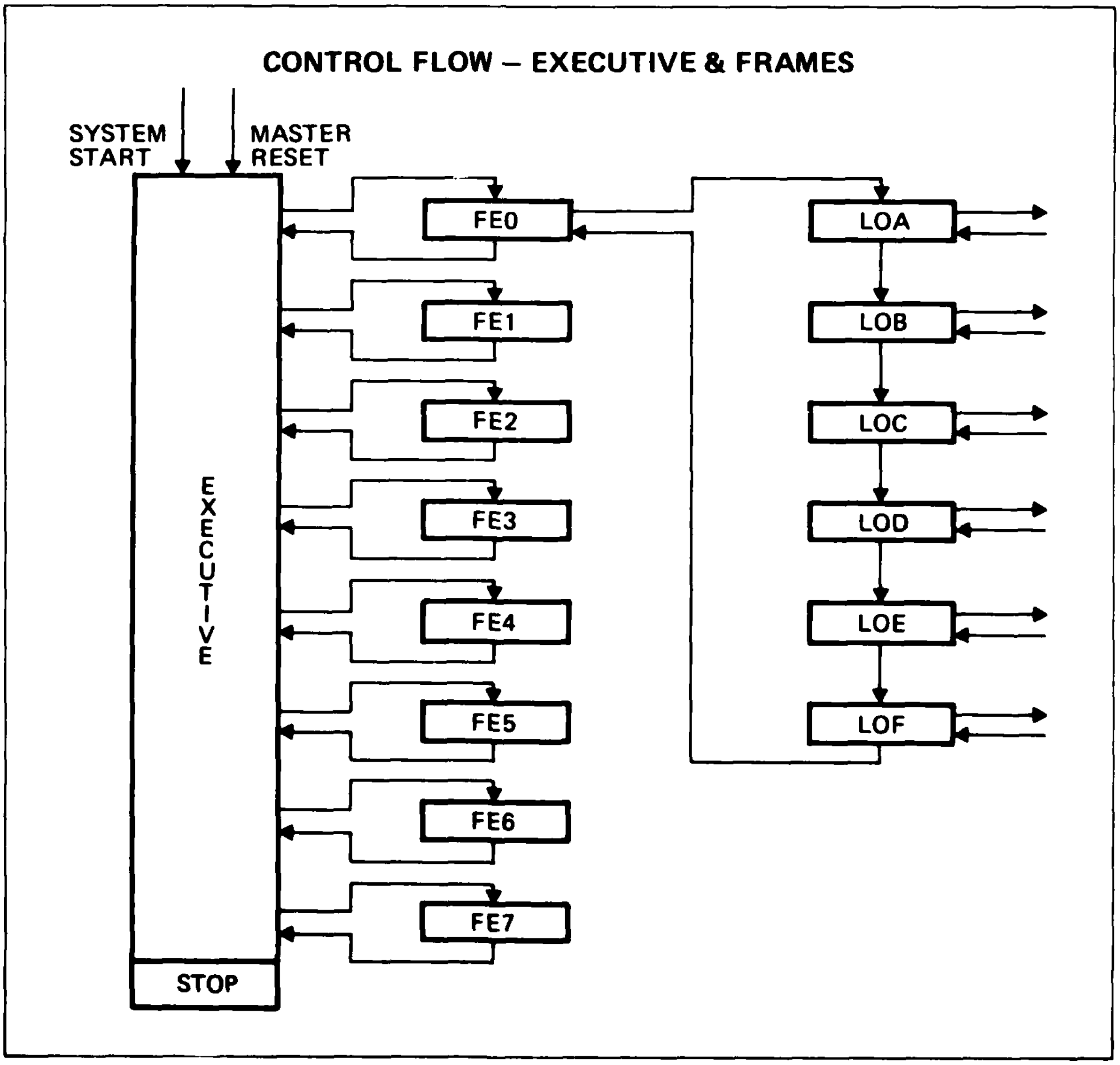

Figure 11 Flight Resident Program Structure

2.3. Flight Software Development Process

The key documents controlling the software design are the Software Requirements Document (SRD) and the Software Structure Document (SSD), prepared in conjunction with BAe from their basic software specification, control law definition and interface documents.

The SRD uses English language and program statements to define the design implementation. These statements are formed to eliminate definition ambiguity and form the basis of definitive software design specifications which are testable to prove the accuracy of the definition.

The SSD defines the running order of the modules within the program segments. The structure is designed to ensure that chronological flow of data from input, through processing, to output is in strict sequence.

An important aspect of the initial software design process is the definition, optimisation and validation of the frequently used algorithms, particularly those associated with system integrity such as signal consolidation and monitoring. MAv developed 6 different voter monitor algorithms to cover the range of analogue and discrete signals at various redundancy levels, together with many other filter and schedule routines.

The codes of practise used in designing and testing the Flight Resident Software FRS are defined in the Programmers Manual and the Testers Manual. These also define the procedures and documentation requirements for configuration and quality assurance control.

The target processor structure, input/output requirements, and the task orientated instruction set, are also rigorously defined.

The overall software development process is shown diagrammatically in Figure 12. The software requirements documents are interpreted to produce software module Design Specifications, which include definition of the module implementation in the form of FORTRAN statements. These high level language statements are then coded into the macro assembler statements used by the FCC processor, supported by FORTRAN comment statements to improve code visibility. A library of well proven macros has been established which

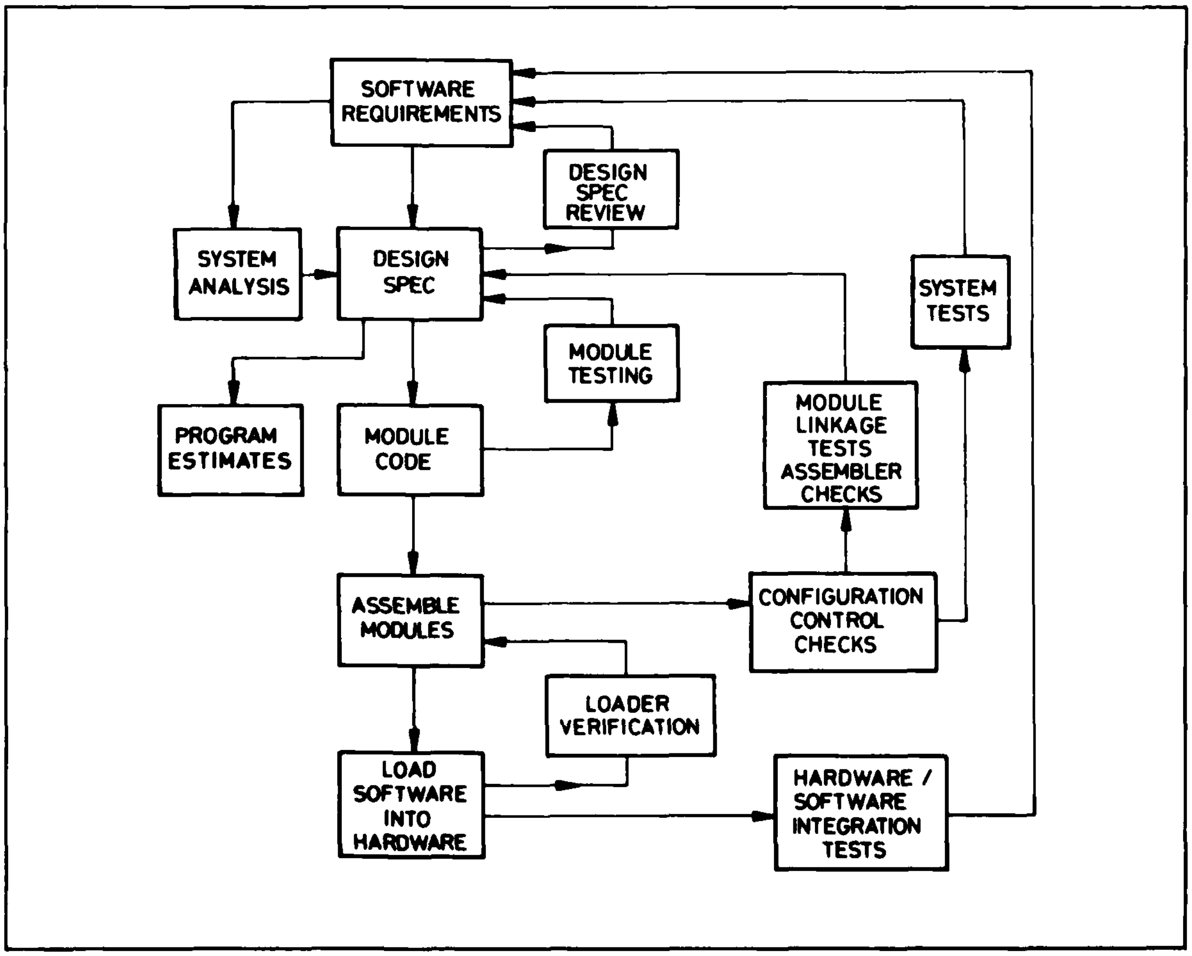

Figure 12 Software Development Process

covers some 70% of the data management and control law software requirements. A corresponding Design Report is produced listing the assembled code for the module together with details including run time, storage requirements, design programmer, module ident, progress card reference, and relevant design calculations. A Module Test Specification is written by another programmer who has neither designed nor coded the relevant module. This procedure minimises the possibility of carrying a module design error through to the test specification. The module code is then tested on a host computer using the specified test harness, and the results are recorded in a Module Test Report.

The module documentation is audited by senior programmers to ensure that the code accurately represents the design requirements and that all design rules such as single entry, single exit, all decision logic in the forward direction etc. have been observed. The audit also ensures that the test rules have been followed including all paths through the module have been exercised and that sufficient intelligent testing has been defined to check overflow/saturation conditions for the module. The test results are correlated with the test specification requirements to ensure all tests are complete and accurate, and the documentation is checked for completeness.

When all the modules are completed the code is assembled into the frames and then the full Flight Resident Software (FRS) with similar testing, reporting and audit at each stage. End to end tests are carried out on the fully assembled programme using the host computer before generation of the PROM device code for transferring the FRS to the target computers. At this stage the Quality Assurance department complete their audit of the software preparation process, check the PROM device code review the design and configuration documentation, and if all is satisfactory release the software for formal issue.

The development and testing of high integrity FRS for Flight Control Systems has been carried out on several host computers using 'in house' developed software tools progressively enhanced, and proven by duplicate assemblies on successive host computers. The result is a suite of well proven support programmes. These programmes include the macro expander, assembler, simulator, PROM code generator and test result annotator. Each includes routines to check valid usage of instructions, storage, work space, run time etc. Any deviation from the rules in these areas inhibits the generation of the final code and the PROM device code.

The SRD, SSD, design and test documents, Programmers and Testers Guides, the generated code and the host computer software are under strict configuration control from the initial issue. Changes can only be introduced by formal Change Requests which are authorised by the Chief Programmer and the Project Manager. Build Standards identify the documentation issues and Change Requests applicable to each issue of the software.

The production of the software is controlled using Progress cards which are identical to those used for controlling manufacture of hardware. These cards create a historical record of all stages of the software development, and the identities of the programmers completing each task. All relevant Change Requests are recorded on the card which can be used to trace the development of the module through all design, test and analysis phases.

Strict adherence to the above techniques generates highly visible FRS, fully audited, well tested and inherently of the required integrity.

3. INTEGRITY APPRAISAL

The complexity, novelty and specified requirements for the IFCS necessitated a major work programme to appraise the resultant integrity. The technique employed analysed the system integrity assuming perfect implementation, and subsequently audited the implementation to assess the effects of possible faults and design defects.

The integrity of the IFCS is primarily determined by the system architecture. Therefore the elements of maximum concern are the points where the redundant lanes are consolidated or otherwise connected, together with the potential for common mode safety critical design defects in the hardware, firmware or software.

The appraisal was carried out using both 'bottom up' and 'top down' analyses, and since some of the issues involved could not lead to useable quantitative estimates of risk, qualitative assessments were also necessary.

The main elements and interactions of the appraisal/audit methodology are shown in Figure 13 and included:-

i) 100% coverage single fault Failure Modes and Effects Analysis (FMEA).
ii) Multiple fault FMEA for specific combinations.
ii) Flight resident software audit.
iv) Appraisal of special areas.
v) Configuration inspection.
vi) Qualification programme.
vii) Burn-in programme.

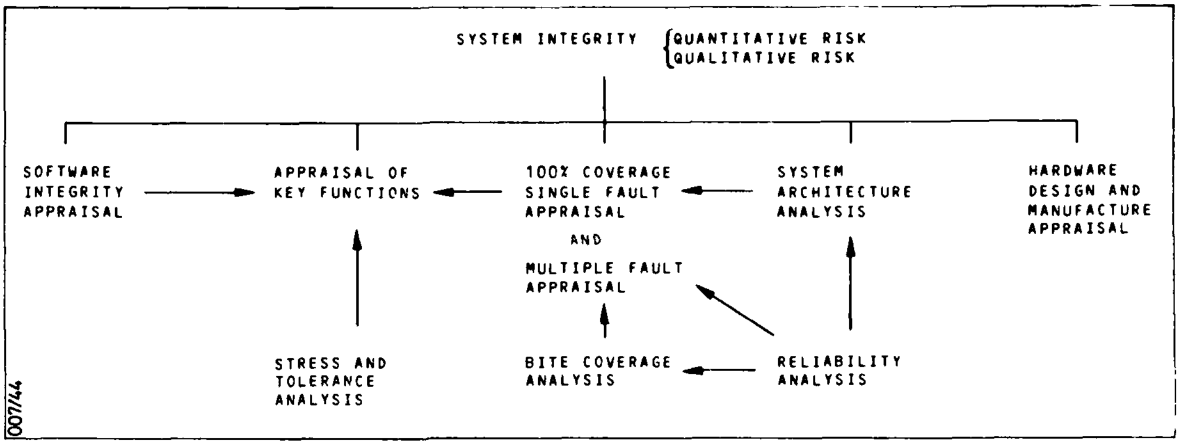

Figure 13 Integrity Appraisal

These primary elements were supported by

a) Module, chassis and LRU FMEA.
b) Microprogram appraisals.
c) Voter/monitor appraisals.
d) Tolerance analyses.
e) BITE coverage analyses.
f) System architecture analyses.
g) Reliability analyses.

During the course of the appraisal detailed technical evaluations of various features and functions of the IFCS were made. The requirements for these evaluations were generated mainly from the FMEA activity, and by BAe as a result of their test activities. These evaluations were reported as a series of Technical Appraisals appended to the overall integrity report, and their results incorporated into the risk assessment.

The integrity appraisal was conducted by a team with specialist knowledge of the equipment design, but to ensure rigour in the appraisal they reported to an independent authority consisting of senior engineers from MAv and BAe.

An essential part of the system clearance depended on the extensive emulator, rig and aircraft testing carried out at BAe, Warton. During these exercises, any unexpected observation that could not immediately be explained by the personnel involved in the test resulted in the raising of a formal query. A written response to every query, approved by both BAe and MAv, was a mandatory requirement for final Q.A. clearance of the aircraft for flight.

A fully detailed description of the system integrity appraisal can be found in Reference 2.

4. FLIGHT TESTING

Following extensive rig and aircraft ground trials, including a considerable amount of electromagnetic compatibility (EMC) and power supply transient testing, the first flight took place on 20th October 1981. Flight testing of the fixed gain control laws was completed in 13 flights, compared with the 14-22 flights budgetted. The aircraft proved easy and straightforward to fly with excellent FCS reliability. The flying rate of the aircraft was never limited by any problems within the FCS but solely by the large amounts of data to be analysed between each flight.

During this 4 month period only one FCS LRU was exchanged due to a defect. The LRU change was prompted, during routine servicing, by BIT detection of a spurious cross lane data transmission malfunction. No in-flight computing malfunctions occurred throughout these trials. A single inflight FCS failure warning occurred just prior to landing on Flight 13, caused by a delay in the quadruplex switch on the undercarriage selector. This switch is a standard Jaguar part, and the possible delay between operation of the two pairs of switch contacts could exceed the time specified in the interface documents.

The FCS detected this delay on a slow undercarriage selection and correctly diagnosed a virtually simultaneous similar double failure resulting in an FCS RED warning to the pilot. However the redundancy management logic successfully dealt with this situation and provided the correct mode selection to the control laws and an otherwise uneventful landing was achieved. After this particular flight, the in-flight BIT failure identification table (FIT) was interrogated via the system Diagnostic and Display Unit (DDU) and immediately identified the cause of the warning. Recurrence of the problem was prevented by a software change to increase the acceptable time delay between the operation of the switch contacts. Pilot confidence in the serviceability of the system

prior to each flight was enhanced by the thoroughness of the BIT function which is a pre-requisite for system engagement. For this demonstrator aircraft, the BIT requires pilot interaction which could be automated to a large extent in a production aircraft environment. However, even this BIT could be completed in about three minutes.

For further details of ground and initial flight testing of the IFCS see reference 3.

5. SOFTWARE REVISION AND FURTHER SYSTEM TESTING

5.1. Scheduled Control Laws for Stable Aircraft

Immediately following certification of the initial issue of FRS, a revision was commenced to incorporate scheduled gain control laws, to enhance the BIT function and to rectify problems encountered during the early trials which had not necessitated immediate correction. This proved to be a very extensive modification exercise resulting in changes to some 75% of the 400 modules comprising the FRS. However the timescale and cost of preparing the new issue was very much less than for the initial issue, and by building on the system integrity appraisal techniques established for the previous system standard, the certification was achieved with less than 20% of the effort required previously. The major changes in system performance required were achieved with only the single hardware modification which changed the contents of the programme store devices.

Recognising the problems of cost, timescale and integrity, associated with software modifications, it was agreed at this stage to be cost effective for additional software segregation to be introduced to the FRS. The 21K words of software required were partitioned across 26K words of store. This was organised not only to provide software segregation at module and segment level, but also to contain different sections of software within separate programme store devices. The objective was to enable future software changes to be contained to a minimum number of software modules and programme store devices. Thus bit for bit comparison of successive FRS assemblies would easily identify the change areas and enable subsequent verification and validation to be more localised than could be justified if the new assembly changed all of the programme store instruction locations.

5.2. Lightning Testing

Lightning protection measures were designed and built into the FCS and it's aircraft interfaces, and extensive EMC susceptibility, bulk current injection and transient testing carried out before the first flight. However, the effects of a lightning strike on an aircraft are unpredictable due to the complex interactive effects of the structure, equipment layout and cable runs. Thus for the early flight trials the aircraft was prohibited from flying in areas where lightning activity was likely. Subsequently a series of simulated whole aircraft lightning tests were carried out to evaluate the effectiveness of the design to protect the FCS from large electromagnetic pulses and thence to obtain a relaxation of the flight restrictions.

In conjunction with the Lightning Studies Unit from Culham (UKAEA) and RAE Farnborough, the tests were carried out by BAe Warton. The simulated lightning pulses were produced by discharging a high voltage, high di/dt generator into the aircraft at the base of the pitot probe. Conductors forming a frame around the aircraft were connected to various parts of the aircraft structure, e.g. tail cone or fin tip, to form the return path for the high current pulses and create the required electric field around the airframe. Extensive monitoring was employed with the measured results being transmitted to the screened recording room via fibre optic data links. Further details of these tests can be found in references 4 and 5.

Test pulses up to 80KV and 100KA were discharged into the aircraft configured into an effectively 'flight-ready' condition, with electrical and hydraulic systems powered and the FCS operating. These pulses represent moderate to severe lightning strikes yet there was no measurable or observable corruption or interference of the FCS function. This has generated considerable confidence in the design techniques used to provide lightning protection for the FCS on the Jaguar aircraft, but extrapolation of the results is necessary to prove the case for rescinding the flight restrictions. MAv are extending these tests by subjecting representative interface circuits to transient voltages defined by Culham as a result of the measurements taken during the whole aircraft lightning tests. These transients are essentially single pulses but with controlled rise, decay and damping characteristics to accurately simulate the extrapolated effects of an extreme lightning strike.

The Jaguar Fly-By-Wire Demonstrator subsequently became the first aircraft to fly after being subjected to whole aircraft simulated lightning tests.

5.3. Flight Test of Scheduled Control Laws

The rig and early aircraft ground trials of the scheduled control laws detected several peculiarities and faults. Intermittent data transmission errors were detected during BIT, and an initially inexplicable incorrect FCS status was occasionally seen at the end of the pre-flight BIT. Several in-flight secondary sensor failures were also recorded.

The majority of these problems were easily identified and diagnosed by use of the BIT and interrogation of the Failure Identification Tables. These were corrected by attention to screening and changes of secondary sensors. However, after several early observations the problem which caused the incorrect post BIT status of the FCS became so infrequent that efforts to capture the history of events leading up to it were unsuccessful. Resolution of the problem prior to commencing the flight testing therefore became dependent on theoretical analysis of the software to predict the possible causes. The structured form of the FRS, and the achieved visibility of the code, enabled the investigation team to establish that there was only one possible way for this situation to develop. Subsequent review of the recorded facts on the incidents, and controlled tests, demonstrated beyond reasonable doubt that this analysis was correct. The situation was caused by occasionally adopting an incorrect procedure that could only be initiated when particular test equipment was connected to the system, and therefore could not occur in flight.

The objective of this phase of flight testing was to assess the aircraft handling with scheduled control laws, check training and spin recovery modes and complete the flutter envelope expansion with a modified standard of tailplane actuator. Testing of the aircraft continued with stores to reduce the manoeuvre margin in preparation for subsequent relaxed stability and unstable flight trials. At the time of writing this paper these trials were approaching a successful conclusion.

5.4. Scheduled Control Laws for Unstable Aircraft

Further revision of the software was required to incorporate the control laws to optimise aircraft performance in the unstable configuration created by addition of ballast and fuel management techniques. This revision required much less change than the previous revision, therefore overall comparison of the tasks cannot be used to assess the benefits obtainable from the introduction of segregation. However at the individual change level, clear benefits have been observed. This is particularly the case for late changes or corrections which could be isolated to a single programme store device change.

Significant reduction in FRS modification time, PROM code generation and hardware reprogramming has been achieved. Combined with increased confidence in the fidelity of the unchanged parts of the programme, these have dramatically reduced the time to introduce and prove late changes immediately prior to the formal validation and verification process. As yet the programme has not reached a stage where formal recertification of the system after a small FRS revision has been attempted. It is not, therefore, possible to state the benefits that segregation provides for this activity, but it is predicted that these could be very significant.

Flight trials of the unstable aircraft control laws are scheduled to commence in June 1983, with aircraft centre of gravity being progressively moved aft to introduce negative static stability.

Further minor changes to the control laws are now being defined to optimise the system for flying the aircraft with the leading edge strakes fitted. These trials should take place later in 1983.

6. EXPERIENCE OBTAINED IN DIGITAL FCS DEVELOPMENT AND CERTIFICATION

The principal aim of the Jaguar Demonstrator aircraft programme has been to establish the feasibility of high integrity digital fly-by-wire systems for future production aircraft, and hence reduce the development timescales and risk for such programmes. In fulfilling this aim, comprehensive development, validation and certification activities have been completed to a depth that has confirmed the major problems and identified practical if not optimum solutions.

The novelty of the system is essentially the use of digital computing therefore the principle experience gained has been associated with software design and certification for very high integrity applications. This is summarised in the following paragraphs.

6.1. Software Requirements Definition

Analysis of the 1300 Change Requests raised during the early phases of FRS development shows nearly half were required because of changes to the specification or misinterpretation of the requirements documents. Significant cost and time savings can therefore be achieved by ensuring an accurate and unambiguous definition of requirements early in the programme. Since some changes of definition are inevitable, particularly for a toally new aircraft programme, structuring and segregation of the software to minimise the rework necessitated by the more probable areas of change also improves the efficiency of producing the FRS.

6.2. Software Segregation and Visibility

Visibility of the FRS structure and code is a pre-requisite to subsequent modification potential, analysis of problems found during system testing and subsequent integrity audit of the software. The production of structured, modular software with stringent

procedural, documentation and configuration control can be tedious and is expensive, but no other technique has yet been established which can enable adequate integrity of the resulting software to be determined.

6.3. Programme Store and Run Time Contingency

Minimising programme store and run time constraints reduces the problems of producing the first issue of a real time software programme. Even greater benefits are found when modifications are subsequently required. Therefore to keep total development costs to an acceptable level, and also maintain visibility of the final software, considerable attention must be given to hardware capability and software design and structure. Cost effective contingency allowances must be made available within the segregated programme store and the segmented software run time structure to allow future modification without the knock on effects of restructuring hardware and/or software or total re-allocation of the programme within the store devices.

6.4. Integrity Audit

The Jaguar Fly-By-Wire programme has developed integrity audit techniques and procedures which have enabled the aircraft to be cleared for flight without having to compromise any of the original requirements. The success of this aspect of the programme has been dependent on many factors including:-

- Independent auditors
- Structuring the integrity analysis to assume perfect implementation, then assessing the probability of defects in the identified critical implementation features.
- Correlation of results from both 'top down' and 'bottom up' analyses
- Constructive use of emulation and control flow analysis techniques
- Dedicating Senior engineering resources to complete a thorough integrity appraisal.

6.5. Development Tools

The task of developing and validating high integrity digital systems can only be achieved in practical timescales if adequate tools are made available. Powerful, efficient and well proven software tools are necessary to contain the task of software production, testing and configuration control. Sophisticated rig facilities are essential to enable thorough testing of the full system executing representative flight tasks in real time. Reliable hardware, with dependable BIT, supported by comprehensive data acquisition and processing facilities enable extensive testing to be carried out in realistic timescales. The hardware and software techniques developed by MAv, complemented by the BAe developed rigs, emulation and data acquisition systems, have identified and assembled a powerful capability for developing future systems.

7. DEVELOPMENTS FOR THE FUTURE

Plans are now being considered for extending the role of the Jaguar Demonstrator aircraft beyond the strakes flight test programme. However any resultant programme is likely to use the aircraft to investigate control techniques rather than concentrate on FCS development. In general, therefore, further software development is expected to be cost constrained to minimum changes within the existing definition, structure and production techniques.

Extensions, adaptations and enhancements of these techniques are therefore being associated with new programmes such as P110/ACA. Building upon the experience established prior to, and during, the Jaguar programme, the following software specification, organisation and coding concepts are now being evaluated.

7.1. Software Requirements Definition

The software requirements definition can introduce problems in three ways - errors, omissions and ambiguities. Improving the methods of definition can do nothing to prevent errors resulting from incorrect assessment of the aircraft characteristics or the control task, but it should be possible to reduce the remaining sources of problems. Most of these are introduced at the boundaries between data bases. Transfer of information from the control law designer, to the requirements documentation, thence to the detail software control specification and eventually the code and test processes, all potentially introduce translation errors, misinterpretations and omissions. Consideration has, therefore, been given to techniques which improve the visibility of these translation processes and provide scope for more automated correlation between the initial requirements and the final code. Writing the initial requirements document in machine executable statements enables the definition to be exercised against the aircraft model, and subsequently the performance of the final code can be checked against the same model. Correlation of the results should then rapidly detect any errors that have been introduced. Adoption of a more 'top down' approach to producing software requirements documents should minimise omissions within the definition and should also provide a more ordered and perhaps more efficient structure.

7.2. Segregation

Extension of the software partitioning already practised can provide further benefits, particularly where the FRS development is to be carried out by more than one organisation e.g. task sharing between avionics supplier and airframe company. As a next step, segregation of the software into two or three essentially autonomous sections is proposed. These would cover for example Executive and Data I/O (type A), Data consolidation and system monitoring (type B) and Control Law tasks (type C). Each would be allocated segments of the programme store and frame run time, with communication via nominated locations within the scratchpad. All work space locations would be read/write protected to minimise illegal data transfer in the event of hardware faults or software design errors. With this structure the software can be developed by separate teams with reduced short term interaction. Since the type A, and to a slightly lesser extent the type B, software will change very little for a given system, the control law changes can be contained within the type C software (perhaps 30% of the programme) with very high confidence that the integrity of the remainder of the programme has not been compromised.

7.3. Task Orientated Programme Language

The standard macro library used for the Jaguar FBW software is being extended to cover the majority of the tasks required by the control law designer. By using macro names and parameters which are familiar to the control law designer, incorporating scaling functions, and providing data fetch and store facilities a task orientated Flight Control Language (FLICOL) has been created. Figure 14 shows an example of a control law path written in this language demonstrating the visibility that can be achieved.

```
    Kr:=0·2
    KXI:=0·6
    KDDay:=0·05
    GET r
    GAIN (Kr)
F10 WASHOUT (3·33)
    SCALE (5·0)
    PUT HOLD 1
    GET XI
    GAIN (KXI)
    SUM (HOLD 1)
    PUT HOLD 2
    GET ay
    GAIN (KDDay)
F11 WASHOUT (20·0)
    SCALE (2·0)
    SUM (HOLD 2)
F12 LAG1 (0·04)
F13 SCHED1 (150·0, 400·0, 1·0, 0·1, 650·0, V)
    PUT ZETA
```

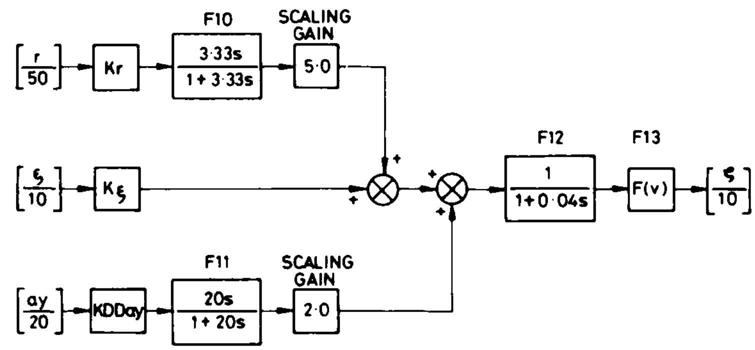

Figure 14 Example of Flight Control Language (FLICOL) Statements

The support tools for this language are based on those presently developed and well proven, providing a relatively simple translation to the selected instruction set of the target processor. These tools can include engineering calculations to relieve the programmer of tasks associated with defining filters, voter monitors, rate limits etc. which are functions of iteration rates.

FLICOL can also be developed as a systems simulation language. This could lead to a situation where the control laws developed on the simulator can be directly translated to the programme for the target processor without the need for source code changes and thus reduces the possibility of introducing errors or misinterpretations.

7.4. High Order Languages

The macro assembler language is considered a highly visible, efficient and safe approach to producing high integrity software, particularly for special purpose processors with instruction sets optimised for flight control applications.

The use of general purpose microprocessors, high order languages and compilers for high integrity applications has caused concern because of the lack of visibility of the device structure, microprogram and compiler 'optimisation' routines. With the development of task orientated microprocessors such as those implementing MIL-STD-1750A, and corresponding languages with more formal verification such as JOVIAL and perhaps ADA these limitations are being minimised. Future use of these, in applications where standardisation of hardware and software production methods are very significant, is being pursued.

ACKNOWLEDGEMENTS

The Jaguar Fly-By-Wire programme has been carried out with the support of the Procurement Executive Ministry of Defence.

The author acknowledges the special team effort that this programme has involved. Success depends upon the combined efforts of British Aerospace, Dowty Boulton Paul and Marconi Avionics. In particular the programme would not have commenced or been able to continue without the enthusiastic support of the British Ministry of Defence and RAE Farnborough. His thanks are extended to colleagues in British Aerospace and the Combat Aircraft Controls Division of Marconi Avionics for assistance provided in the preparation of this paper.

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L'INTERFACE HOMME - MACHINE
dans les avions commerciaux de la nouvelle génération

par

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INTRODUCTION

Le but du présent exposé est de présenter, après un bref rappel historique de ce qu'a été l'interface homme-machine depuis les tout débuts de l'aviation, les diverses tendances de cet interface pour la dernière décennie du siècle.

Un simple coup d'oeil, même non averti, ne peut que constater une profonde évolution dans le style des rapports entre les aéronefs et leurs équipages. Ce que l'on voit peut-être moins, c'est que cette évolution a procédé par bonds.

C'est un bond spectaculaire que l'interface homme-machine est en train de franchir au milieu des années quatre vingts. On se propose au cours de cet exposé, volontairement succinct, de mettre en évidence les caractéristiques profondes de ce bond.

1. HISTORIQUE DE L'INTERFACE HOMME - MACHINE

1.1. L'INTERFACE DES PREMIERS TEMPS

Après une phase fugitive pendant laquelle l'homme a volé en se servant de ses seuls sens (vue, ouïe, *sensations musculaires*), non sans difficulté car ces sens n'étaient pas adaptés à ce nouvel environnement, on a imaginé de traduire les paramètres du vol en valeurs numériques les imageant de façon intelligible.

Ces premières valeurs numériques étaient élaborées en des positions de l'avion directement accessibles à l'oeil du pilote.

Le premier interface homme-machine était constitué par exemple :

- d'une colonne de liquide se déplaçant sous l'action de la dépression d'un tube venturi, et donnant une idée de la "vitesse" grâce à une loi approchée de M. BERNOULLI;

- de repères sur la structure au droit des parties mobiles;

- de "jaugeurs" à lecture directe, par hauteur du carburant dans des tubes transparents communiquant avec les réservoirs.

1.2. LES CHAINES DE MESURE INDIVIDUELLES

Très vite, il est apparu que les paramètres dont la variation permettait au pilote de gérer au mieux son vol et la machine variaient dans des milieux éloignés du poste de pilotage lui-même: pression d'huile dans les moteurs, quantité de carburant dans les réservoirs d'aile, position des compensateurs en extrémité de voilure, etc ....

Il a fallu concevoir les premières "chaînes de mesure", qui se sont vite généralisées sur les avions, selon le schéma suivant :

capteur "in situ" ⟶ ligne de transmission ⟶ instrument dans le cockpit.

Ce développement coïncidant avec une meilleure maîtrise de l'énergie électrique, très vite on a traduit le paramètre à mesurer en signal électrique (signal analogique variable en intensité, tension ou phase) facile à faire cheminer jusqu'au cockpit dans un fil électrique qu'il était aisé de ranger dans l'avion.

On a vu apparaître de multiples chaînes de mesure du type :

capteur de position (montage en pont de Wheatstone)	—— fil ——▶	indicateur de position au tableau de bord
capteur radioélectrique (récepteur ADF à cadre asservi)	—— fil ——▶	indicateur de gisement (recopie de position du cadre)

Avec, en variante, des transmissions de pression au travers d'un tube cheminant directement d'un capteur de pression jusqu'au tableau de bord.

1.3. LA GENERATION DES CENTRALES

Dans le courant des années soixante, la masse des informations traitée dans le cockpit devenait extrêmement importante, et l'on avait affaire à des paramètres qu'il fallait à la fois diffuser en différents endroits, et utiliser sous forme de facteurs correctifs pour l'élaboration d'autres paramètres.

Ainsi la pression statique, envoyée dans des instruments primaires (altimètres), servant à élaborer d'autres paramètres (la "vitesse anémométrique" dans les anémomètres), et intervenant dans des automatismes comme les régulateurs de moteurs ou les systèmes de pressurisation.

Pour éviter de multiplier à l'infini les lignes de transmission (dont la longueur finissait par se mesurer en dizaines de kilomètres sur un avion de ligne), on en est arrivé concevoir des "centrales": l'avion-type de la fin de cette décennie comportait par exemple :

- une centrale anémométrique
- une centrale gyroscopique
- une centrale de cap
- une centrale de navigation.

Chaque "centrale" était un calculateur analogique qui :

- d'une part recevait directement de capteurs élémentaires des paramètres de base "bruts" (par exemple pressions statique et totale, température d'impact pour une centrale anémométrique);

- d'autre part distribuait vers divers "utilisateurs" (instruments du tableau de bord ou systèmes divers) des paramètres élaborés (température statique, vitesse propre, nombre de Mach).

1.4. LA NOUVELLE GENERATION

Le milieu des années soixante-dix a vu l'apparition de progrès techniques divers et décisifs :

- des calculateurs numériques à faibles masse et encombrement, à grande capacité de calcul (permettant le calcul en temps réel de paramètres multiples, complexes et à variation rapide), et à grande capacité de mémoire (permettant le stockage de données liées au domaine de vol ou à l'aire géographique de travail);

- des tubes cathodiques couleurs à haute définition et grande fiabilité (technique dite "shadow-mask");

- des systèmes de collimation lumineux, clairs et peu encombrants.

Ces progrès ont permis une nouvelle conception de l'interface homme-machine. L'avion des années quatre vingts peut être présenté sous la forme schématique suivante :

- au centre de l'avion, le coeur de tous les systèmes, un calculateur numérique central quasi-invulnérable (haute redondance dans les calculs, protection totale contre les perturbations extérieures), à capacité de calcul et de mémoires et à vitesse de travail permettant la gestion en temps réel de tous les systèmes et de l'ensemble du vol;

- en amont, des capteurs divers et variés placés aux endroits où doivent être faits les mesures brutes, avec conversion analogique-numérique des paramètres sur place, et transfert sous forme numérique (inaltérable) au calculateur central;

- en aval, des appareillages divers, tubes cathodiques de planche de bord essentiellement, collimateurs de pilotage, instruments isolés, sous-ensembles techniques comme les commandes de vol électriques;

- pour le dialogue direct enfin entre l'équipage et le calculateur (dans les deux sens, ordres et questions, introduction de données dans le sens homme-machine et réponses dans le sens machine-homme) des unités de dialogues dites "C.D.U." (command/display units) composées d'un clavier de touches de fonction et alphanumériques et d'un tube cathodique à lecture alphanumérique.

Bien entendu, le schéma simplifié ci-dessus n'est pas apparu globalement dans son ensemble. Les réalisations du début des années quatre-vingts sont partielles: les tubes cathodiques de planche de bord sont doublés d'instruments individuels "de secours", les commandes électriques sont doublées de "secours" mécaniques (servo-tabs).

Mais la tendance, à l'horizon quatre-vingt-dix, est celle ci-dessus décrite.

Dans la suite de l'exposé sont abordés tour à tour les différents composants de l'interface de la fin du siècle. On y montrera pour chacun l'état des études, les tendances et les réalisations.

2. LES TUBES CATHODIQUES DE PILOTAGE

2.1. TECHNIQUE ET PREMIERES REALISATIONS.

L'idée de base ayant conduit à l'apparition des premiers tubes cathodiques de pilotage est qu'il est à la fois plus fiable et plus souple de travailler sur de l'électronique que sur du mécanique.

On a donc imaginé de remplacer les instruments électro-mécaniques de pilotage par des tubes cathodiques à balayage (sortes de petits postes de télévision installés sur la planche de bord).

Pour que ceci soit réalisable, il a fallu sur le plan technique résoudre un certain nombre de problèmes, qu'on cite pour mémoire :

- obtenir une très grande finesse de dessin (par augmentation du nombre de lignes de balayage);

- pouvoir jouer sur de nombreuses couleurs bien individualisées (on arrive aujourd'hui à une dizaine de couleurs) conservant leur contraste relatif en cas de panne d'un des canons monochromatiques de base;

- avoir une luminosité apparente constante, indépendante des conditions ambiantes (que le tube soit dans l'ombre ou au soleil);

- avoir une cadence de renouvellement de l'image suffisamment élevée pour que les parties mobiles évoluent de façon lisible pour l'oeil (ce qui est obtenu par des fréquences de rafraîchissement de 50 Hertz ou plus).

La technique étant au point, on a dans un premier temps recopié purement et simplement les instruments de vol principaux (ADI ou "attitude director indicator" et HSI ou "horizontal situation indicator").

C'est ce qui a été fait par exemple sur l'ADI cathodique des EADI/HSI (tubes cathodiques de pilotage) des B.757 et B. 767 (avec maquette, horizon, directeur de vol, "fast-slow", écarts loc et glide) ou sur le HSI des EFIS (tubes cathodiques de pilotage) de l'A.310 , en mode "rose" (rose des caps, repère haut de cap, écart latéral VOR ou LOC , aiguilles ADF).

2.2. CONCEPTION D'UNE NOUVELLE SYMBOLOGIE

Les immenses possibilités offertes par les tubes cathodiques ont permis d'envisager des symbologies entièrement "repensées", qui auraient été impossibles sur des instruments électromécaniques classiques.

La tendance est celle d'une concentration des informations sur un tube unique. Ainsi :

- sur l'ADI cathodique des EFIS de l'A.310, on a regroupé, autour de l'horizon classique et de son directeur de vol, à gauche un anémomètre évolué (plus "riche" qu'un anémomètre classique), à droite une information (sommaire) d'écart d'altitude;

- sur les HSI cathodiques des EADI/HSI des B.757/B.767 et des EFIS de l'A.310 on a fait apparaitre des "cartes" (avec tracé de l'itinéraire prévu, dessin de circuits d'attente, localisation d'aides radio ou d'aérodromes) avec, en surimpression, des informations (synthétiques) issues du radar météorologique.

Mais la concentration n'est pas le seul résultat auquel on a pu arriver grâce aux tubes cathodiques de pilotage. On a pu faire apparaître des informations absolument inédites dont on cite ci-dessous quelques-unes parmi les plus intéressantes :

- anémomètre linéaire (EFIS A.310) équipé des limites de domaine actuelles réelles élaborées par le calculateur central, haute (fonction de la configuration actuelle) et basse (fonction de l'incidence actuelle) et d'une "prévision" de vitesse à 10 secondes (élaborée à partir de l'énergie totale actuelle);

- horizon artificiel (EFIS A.310) à allègement de symbologie alertant à l'approche d'attitudes anormales;

- accès plus direct à la trajectoire par représentation sur l'ADI (EFIS A.310) d'un "vecteur-vitesse sol" permettant de suivre en approche, en l'absence de moyens radioélectriques performants, une pente sol et une route vraie constantes, indépendamment des conditions extérieures (variations de vent en particulier);

- indicateur de dérapage électronique couplé à l'indicateur d'inclinaison latérale (EFIS A.310).

- indicateurs tridimensionnels (à l'étude) d'approche MLS (études du système "VEGA" et associées au Centre d'Essais en Vol de Brétigny, Etudes de la NASA à Ames).

2.3. LE TOUT-CATHODIQUE DE PLANCHE DE BORD

Certains concepteurs vont plus loin et, suivant en cela certains projets avancés (parmi lesquels on peut citer le JAS suédois de SAAB), envisagent de recouvrir la totalité de la planche de bord par des écrans de télévision à grande surface portant la totalité des informations nécessaires au pilotage.

On peut citer parmi les études en cours pour l'aviation commerciale les tubes cathodiques intégrés à grande surface de Smiths, qui volent depuis plus d'un an sur un BAC.111 du RAE en Grande-Bretagne.

Le virage vers de tels tableaux de bord n'est pas définitivement pris, mais les orientations prises par les secteurs classiquement de pointe en ce domaine (on peut ici citer la navette spatiale) laissent à penser que l'avenir des tubes cathodiques de planche de bord reste entier dans l'aviation commerciale.

2.4. LES PROBLEMES POSES PAR LES TUBES CATHODIQUES DE PILOTAGE.

Les tubes cathodiques de pilotage sont déjà en service sur des avions commerciaux certifiés de la "grande" aviation (B.757/767, A.310) comme de l'aviation d'affaires.

Pourtant, tous les problèmes ne sont pas résolus pour autant, et les services officiels des différents pays concernés ont à se pencher d'urgence sur certains problèmes essentiels qu'il conviendra de régler rapidement sous peine d'aboutir à des situations inextricables :

- mise au point d'une réglementation internationale (FAR ou JAR) fixant les règles de conception et de réalisation des tubes de pilotage, qui n'en est encore qu'au stade des études préparatoires;

- accord sur une semi-standardisation sur les symboles (en particulier pour les paramètres nouvellement présentés) voire sur les couleurs et d'autres principes de figuration (symboles momentanément représentés en tiretés, ou clignotants);

- accord sur une nouvelle conception globale de la planche de bord - c'est-à-dire sur les positions relatives des nouveaux symboles sur les nouveaux tubes de pilotage, et de ces tubes de pilotage eux-mêmes, comme on l'avait fait pour les instruments classiques ("T" de planche de bord pour les quatre instruments de base).

3. LES TUBES CATHODIQUES DE GESTION DE SYSTEMES

3.1. TUBES CATHODIQUES DE CONTROLE MOTEURS

De même qu'il a été vite tentant de remplacer des instruments de vol électro-mécaniques par leur recopie sur des tubes cathodiques couleurs, de même on a rapidement envisagé, avec l'avènement de ces tubes perfectionnés, de remplacer les instruments de contrôle des moteurs par des tubes cathodiques les recopiant.

C'est ce qui a été fait en version de base sur les B.757/767, où des tubes cathodiques, installés au centre de la planche de bord, entre les deux pilotes, reproduisent les instruments classiques de contrôle des moteurs.

En fait, ces tubes apportent sur les instruments électro-mécaniques plusieurs avantages sur le plan de l'interface :

- l'aspect analogique est conservé en s'allégeant (une "aiguille" se déplace devant des secteurs colorés classiques vert-ambre-rouge, dépouillés de graduations et de chiffres);

- une seule valeur numérique apparaît, la valeur actuelle, qui a elle-même une couleur codée (vert en échelle d'utilisation normale, ambre en zone marginale et rouge en secteur dangereux), avec d'autres avatars possibles (augmentation de taille, clignotement).

Il reste que ce type d'information sur un tube cathodique, en amont duquel on dispose de calculateurs performants, reste très en-deçà des possibilités actuelles.

On donne ci-dessous deux exemples de tubes cathodiques associés à des programmes de gestion de systèmes :

- les tubes descriptifs de systèmes
- les tubes de guidage d'action (tubes de présentation de "do-lists" ou de "check-lists").

3.2. TUBES DESCRIPTIFS DE SYSTEMES

Un des deux tubes du système de gestion des systèmes de l'A.310 (ECAM), conçu et réalisé par THOMSON-CSF présente en permanence un des circuits essentiels de l'avion (par exemple: circuits freinage, hydraulique, électrique, carburant, etc ....)

Chaque circuit est représenté de façon assez simplifiée, mais de façon à faire apparaître les "lignes" essentielles, avec un code couleurs (les circuits en service, sous tension ou sous pression, étant représentés d'une couleur différente de celle des circuits isolés), les vannes (ou contacteurs, robinets disjoncteurs, etc ...) avec les représentations ouvert/fermé et certaines valeurs numériques (et parfois aussi analogiques) en certains points clés.

Un seul coup d'oeil sur chaque circuit permet de déceler les défaillances éventuelles qui le pénalisent, et on peut rapidement y lire le résultat d'opérations qu'on y fait par ailleurs (la fermeture volontaire d'une vanne y est "vue" instantanément, avec l'isolement des parties de circuit qu'elle provoque).

Pour chaque circuit, une figuration unique sur le tube cathodique permet de remplacer avantageusement la consultation de nombreux instruments classiques.

Comme dans ce type de tube un seul circuit peut être représenté à la fois, il importe de définir une logique d'apparition des circuits sur le tube unique. Cette logique, pour les ECAM de l'A.310 est la suivante :

- en vol normal, le circuit présenté est celui qui est le mieux adapté à la phase du vol en cours (circuits moteurs à la mise en route, circuit freinage au roulage, circuit de pressurisation - climatisation en croisière);

- l'équipage conserve à chaque instant la possibilité d'appeler le circuit qu'il désire consulter;

- en cas de panne sur un circuit, le circuit concerné est présenté en priorité (avec une présentation séquentielle basée sur les urgences respectives en cas de panne multiple).

3.3. TUBES "DO-LIST" OU "CHECK-LIST"

Un tube cathodique subordonné au calculateur (ou au programme) de gestion des systèmes peut être réservé à la présentation d'une liste d'opérations à effectuer ("do-list") ou à contrôler ("check-list") par l'équipage.

La présentation de telles listes sur tube cathodique présente de nombreux avantages par rapport aux check ou do-lists papier :

- elles sont appelées automatiquement en fonction de la phase du vol en cours (en conditions normales) ou de la panne (en conditions anormales) - avec enchaînement relatif des deux dans les cas critiques (feu au décollage par exemple);

- le graphisme (taille des lettres , souligné) ou la couleur peuvent attirer l'attention sur les actions les plus urgentes;

- un code couleur particulier peut permettre le contrôle par le système de la bonne exécution des opérations prescrites par l'équipage: si la "do-list" prévoit sur une ligne l'ouverture d'un robinet d'intercommunication, la couleur de la ligne ne changera (pour prendre la couleur "opération effectuée") que lorsque le calculateur de gestion aura effectivement reçu d'un capteur l'information que le robinet a bien été ouvert. D'où une sécurité accrue.

4. LES SYSTEMES DE GESTION DU VOL

4.1. LE PROBLEME DE LA GESTION DU VOL

Gérer un vol commercial de façon efficace, c'est le conduire d'un point à un autre dans les meilleures conditions de sécurité et d'économie.

Dans la pratique, cela implique à la fois la gestion optimisée d'une trajectoire et celle du carburant, avec les interférences permanentes qu'il y a entre cex deux gestions.

Dans les premiers temps de l'aviation commerciale, l'équipage avait à assurer la totalité du travail de gestion. Pour prendre un exemple seulement dans la gestion de la trajectoire, il ne disposait que des valeurs brutes de paramètres isolés (cap magnétique, vitesse anémométrique, relèvements radioélectriques), à partir desquels il devait déterminer continuellement sa position, et définir les corrections propres à le rapprocher de sa trajectoire idéale.

Plus tard, des calculateurs analogiques aux performances limitées ont permis d'alléger son travail en calculant des paramètres plus élaborés (vitesses propre et sol, route vraie, position radioélectrique en radial/distance).

L'avènement des calculateurs numériques à grande capacité de calcul et de mémoire a permis, dès le début des années soixante-dix de résoudre le problème de la gestion de la trajectoire en plan (guidage automatique de l'avion sur une trajectoire prédéterminée à deux dimensions).

L'évolution rapide des performances de ces calculateurs a permis d'envisager la gestion de la trajectoire à trois dimensions (profil de vol vertical en plus), puis quatre dimensions (avec la variable temps), et d'y intégrer enfin la gestion optimisée du carburant.

Ces différents systèmes de gestion du vol sont abordés tour à tour ci-après.

4.2. LES SYSTEMES DE NAVIGATION DE ZONE

Les premiers systèmes de gestion du vol à avoir été opérationnels ont été les systèmes de gestion de la navigation.

Un bon exemple en est l'"Areanav" de Collins,proposé en option sur le DC.10 dès 1970. Le principe en est le suivant :

- un calculateur central dispose en mémoire de toutes les informations de navigation et de circulation aérienne concernant la région couverte par les vols de la Compagnie;

- il détermine à chaque instant la position de l'avion, à partir d'une position "estimée" moyenne des positions de trois centrales inertielles) et de recalages radioélectriques sur des VOR/DME (qu'il affiche lui-même parmi les mieux appropriés grâce à une fonction dite "auto-tune ");

- il guide directement l'avion (au travers du pilote automatique) sur la trajectoire du vol, prédéterminée et affichée par l'équipage avant le départ;

- il fournit à l'équipage tous les renseignements de navigation annexes que celui-ci peut désirer (distances, heures de survol) au travers d'une unité de dialogue ("C.D.U.") classique.

4.3. LES SYSTEMES DE GESTION DE PERFORMANCES

Le premier "choc pétrolier" de 1973 étant contemporain de l'essor des calculateurs numériques embarqués, de nombreuses sociétés se sont mises à étudier des systèmes de gestion de performances (ou "P.M.S.", performance management systems).

- soit en rattrapage sur des avions déjà en service (par exemple Lear-Siegler pour les B.727/737, BOEING-DELCO pour le B.747, SFENA pour l'A.300);

- soit un système de base pour des avions nouveaux de classe intermédiaire (les avions nouveaux de "premier niveau" ayant été directement bénéficiaires des "F.M.S." étudiés plus loin).

Ces systèmes supposent résolu le problème de la trajectoire en plan et calculent, à partir des informations de masse avion, performances potentielles des moteurs et paramètres extérieurs :

- le carburant minimal à embarquer avant le décollage;
- le profil vertical idéal de vol (montée, altitude de croisière, point de début de descente).

Ces systèmes peuvent en général être couplés au pilote automatique et aux automanettes, et gérer directement la trajectoire dans le plan vertical.

Ils ont leur interface avec l'équipage assuré par un C.D.U. particulier, dont les touches de fonction et les pages sont adaptées aux fonctions particulières du système.

4.4. LES SYSTEMES COMPLETS DE GESTION DU VOL

Les avions commerciaux de la toute dernière génération, dont les premiers ont été certifiés au début de 1983, ont été dès le départ conçus avec à leur bord un système de gestion globale du vol ("F.M.S." ou flight management system): le F.M.S. de SPERRY (pour l'A.310 et les B.757/767), le F.M.S. de SMITHS (pour l'A.310).

En fait, dès 1977 un F.M.S. (conçu par LOCKHEED-ARMA) était proposé en option sur les L. 1011 Tristar.

Il n'y a pas grand'chose à ajouter sur les F.M.S. quand on a dit que ce sont des systèmes qui, au travers d'un C.D.U. unique et relativement complexe, tant pour son clavier que pour l'information diffusée dans ses "pages") assurent la gestion globale du vol, avec les capacités rajoutées des systèmes de navigation de zone et des P.M.S. présentés précédemment.

En résumé, le F.M.S. :

- détermine à chaque instant une position précise de l'avion avec une précision de l'ordre de 1/10è de NM en zone ouverte radio-électrique);

- détermine la trajectoire optimale dans le plan vertical;

- guide directement l'avion dans les trois dimensions par le biais du pilote automatique et des automanettes;

- effectue tous les calculs concernant la navigation (distances, heures de survol) et la gestion du carburant (calculs avant vol, calculs d'autonomie et d'attente, voire de dégagement en vol).

Dans leur définition actuelle et, a fortiori, dans les versions encore améliorées qui verront le jour dans les années à venir (et qui sont déjà à l'étude), les F.M.S. permettent :

- d'augmenter la sécurité en dégageant l'équipage d'une charge de travail que les calculateurs peuvent assumer mieux que lui, et en le laissant disponible pour traiter des problèmes graves en cas de nécessité (il est d'ailleurs symptomatique que - en dehors de toute polémique - l'apparition des F.M.S. corresponde à une réduction de l'équipage minimal de conduite sur les avions de premier niveau);

- d'augmenter la rentabilité par une optimisation de la trajectoire dans toutes les dimensions, avec les économies en carburant qui en résultent.

5. LES VISEURS DE PILOTAGE

5.1. PRINCIPE DE BASE DES VISEURS DE PILOTAGE

Une analyse détaillée du comportement d'un pilote dans les phases d'évolution au voisinage du sol (décollage et, surtout, atterrissage) montre qu'un temps considérable est perdu pour le transfert de l'attention entre :

- le monde extérieur d'une part, vu au travers du pare-brise, monde vis-à-vis duquel, en l'état actuel des choses, commence et s'achève tout vol par un contact visuel direct,

- les instruments de pilotage d'autre part, permettant d'avoir accès au travers de paramètres et d'un certain symbolisme à une trajectoire non directement accessible.

Bien que ces phases de décollage et d'atterrissage ne constituent en pourcentage de temps qu'une part infime du vol, leur importance reste primordiale. Elles sont en particulier le siège de l'écrasante majorité des accidents aériens, et sont de ce fait à traiter avec une attention particulière.

Une manière élégante de diminuer la perte de temps en transfert de l'attention consiste à faire apparaître dans le champ visuel tous les éléments nécessaires à une bonne appréhension de la trajectoire. L'oeil du pilote n'a plus alors qu'un objectif sur lequel accommoder: le monde extérieur, sur lequel viennent se superposer les informations appropriées.

Ceci est possible techniquement au travers de collimateurs (ou "H.U.D.", Head up displays) dont la définition peut être ainsi faite:

- une glace semi réfléchissante, située au voisinage du pare-brise, laisse au pilote la vue totale sur le monde extérieur;

- un calculateur élabore, au travers d'un générateur de symboles, des signaux lumineux (symboles analogiques ou informations numériques) qui sont projetés à l'infini en face du pilote, et viennent se superposer pour lui au monde extérieur, où ils sont "perçus" ou "lus" sans accommodation particulière.

A partir de ce principe, et une fois les difficultés techniques surmontées (champ assez vaste, obtenu par les nouveaux viseurs "à diffraction", sécurité, obtenue par des glaces facilement effaçables, voire des projections directement dans le pare-brise, finesse de la symbologie, adaptation de la luminosité, couleurs, etc ..) diverses solutions pratiques sont possibles.

On en donne seulement ci-dessous deux grands types, pour montrer les directions dans lesquelles peut se développer l'évolution des viseurs de pilotage.

5.2. LES VISEURS "ÉLÉMENTAIRES"

Dans ces premiers viseurs, dont le développement a commencé très tôt en France (dès la fin des années soixante, programme ERA au Centre d'Essais en Vol de BRETIGNY), on a tenté de fournir au pilote des paramètres élémentaires, élaborés à partir de capteurs simples, et présentés sous forme analogique plus immédiatement utilisable en pilotage-réflexe.

Ces premiers viseurs ont permis de repenser d'une certaine manière le problème du pilotage, en fournissant au pilote des paramètres en prise directe sur la trajectoire (incidence, vecteur vitesse, pente totale) à la place de paramètres classiques cernant cette cette trajectoire de façon plus indirecte ("vitesse" anémométrique, assiette, cap, vitesse verticale, paramètres figuratifs de la poussée des moteurs).

Sur le plan de l'aéronautique civile commerciale, le premier viseur de cette tendance à avoir été commercialisé a été le viseur C.S.F. 193 qui, installé sur l'avion Mercure, a permis dès le début des années soixante-dix d'en abaisser de façon spectaculaire les minimums d'approche (en approche de précision de catégorie III) par l'aide qu'il apportait à la transition vol aux instruments - vol à vue.

Actuellement, le viseur de cette tendance le plus significatif, bien qu'il n'ait pas été commercialisé (à l'encontre d'un de ses homologues installé sur Super-Etendard) est le THOMSON-CSF TC.125 ERA. De nombreuses campagnes d'évaluation, ouvertes à de nombreux constructeurs, utilisateurs et services officiels, ont mis en évidence les performances particulièrement intéressantes de ce viseur.

Ses caractéristiques sommaires sont les suivantes :

- figuration de paramètres élémentaires directs de pilotage (incidence, vecteur vitesse, pente totale) élaborés à partir de moyens simples (sonde d'incidence, accéléromètres, références inertielles);

- figuration d'une piste synthétique qui vient de superposer à la vraie piste en vol à vue, à partir des signaux ILS ("lissés" par inertie);

- pilotage au travers de boucles logiques raccourcies dans lesquelles le pilote reste un élément essentiel;

- technique de vol plus simple, identique en conditions de vol à vue (VMC) et de vol aux instruments (IMC);

- meilleure réponse, grâce à des temps de réponse plus courts, en cas de perturbations externes (gradient de vent) ou internes (perte d'un moteur);

- remarquable adaptation à la transition vol aux instruments-vol à vue, comme il a été dit.

5.3. LES VISEURS "DIRECTIFS"

Bien que les viseurs "élémentaires" ci-dessus présentés puissent apporter de profonds avantages par leur principe même, ce ne sont pas forcément eux qui s'imposeront les premiers dans le domaine de l'aviation commerciale.

D'autres viseurs ont suivi un développement parallèle, en particulier sous les auspices de la NASA, et ont abouti à des réalisations commerciales, comme le viseur Sundstrand proposé en option sur le DC.9 - 80 (hélas pénalisé par son absence de référence inertielle).

Ces viseurs, dont le viseur A.M.D. PERSEPOLIS (proche du viseur du Mirage 2000 en ce qui concerne les fonctions pilotage) est un des plus significatifs, peuvent avoir les caractéristiques suivantes :

- figuration de paramètres de vol plus "classiques", assiette et "vitesse" anémométrique par exemple;

- figuration d'un vecteur-vitesse "sol", moins facile d'élaboration que le vecteur-vitesse "air" des viseurs "élémentaires", et plus attrayant pour l'utilisateur même si parfois moins fiable;

- présentation d'un "directeur de vol", sorte de "boîte" dans laquelle il convient de piloter le vecteur-vitesse pour rejoindre et suivre une trajectoire nominale - au prix de l'élimination du pilote de la boucle de pilotage où un calculateur le remplace;

- présentation d'un "directeur de poussée", sorte de "boîte" dans laquelle il convient de "piloter" les repères d'énergie totale de l'avion, en s'aidant des éléments agissant sur le bilan poussée-trainée, manettes des gaz en particulier;

- élaboration d'une piste synthétique utile à la transition vol aux instruments-vol à vue.

5.4. LES VISEURS DE LA FIN DU SIECLE

Contrairement aux autres composants de l'avion commercial "de l'an 2000", qui progressivement se mettent en place sur les nouveaux produits de l'industrie aéronautique, les viseurs, tout en continuant à progresser dans le domaine de l'étude (surtout en France, au Centre d'Essais en Vol, et aux Etats-Unis, à la NASA, marquent le pas dans le domaine des réalisations pratiques.

Cela est dû à plusieurs raisons, en particulier à la nécessité pour les constructeurs d'amortir de coûteuses études sur les pilotes automatiques (qui semblent dans un premier temps concurrencer les viseurs), à l'abaissement relatif du coût desdits pilotes automatiques avec l'avènement du numérique, à la réticence enfin d'utilisateurs assez conventionnels à des pas en avant trop réformateurs.

Quoi qu'il en soit, et l'exemple de la navette spatiale n'en est qu'un exemple parmi d'autres, les viseurs de pilotage auront probablement une place essentielle dans l'interface homme-machine de l'avion commercial de la fin du siècle - bien qu'il soit difficile d'en prévoir aujourd'hui leur forme exacte.

6. AUTRES ASPECTS DE L'INTERFACE DE DEMAIN

6.1. LES POSTES DE COMMANDE MULTIPLEXÉS.

En plus des grandes tendances présentées plus haut, l'interface homme-machine des avions commerciaux de la prochaine génération va présenter des nouveautés qui, pour être moins importantes, n'en auront pas moins des retombées qui modifieront profondément l'environnement classique.

Parmi elles, les postes de commande multiplexés.

Le principe de ces postes, dont certains sont déjà en service de façon limitée dans l'aviation d'affaires, et les plus évolués en essais, est le suivant :

Les cockpits actuels sont encombrés de poste de commande divers dont la manipulation n'intervient que pendant une partie très limitée, pour ne pas dire négligeable, du vol. Par exemple, sur un avion équipé de trois postes V.H.F., le V.H.F.2 utilisé pour veiller la fréquence de détresse 121.5 MHz ne sera pas manipulé de tout le vol alors qu'il occupe une place non négligeable dans le cockpit.

On a donc imaginé des postes de commande, qui ne sont autres que des C.D.U. particuliers, qui permettent de s'adresser,au travers d'un programme simple, à différents postes concernant les communications (V.H.F., H.F.), les moyens de navigation (VOR, DME, ADF, INS) et de guidage (ILS, MLS), les moyens de visualisation (radar, HUD) etc...

Lorsqu'un des membres d'équipage veut modifier un affichage (par exemple la fréquence du poste HF.2), il "appelle ce poste" à son CDU (il y a un C.D.U. de ce type par pilote), vérifie la fréquence en service (et éventuellement la fréquence affichée en réserve), et procède par frappe (au clavier alphanumérique) et insertion pour afficher une nouvelle fréquence.

Le C.D.U. du poste de commande multiplexé s'adressant à un calculateur équipé de mémoire, on peut même imaginer, pour éliminer des erreurs potentielles, de faire afficher par le membre d'équipage-opérateur :

- un indicatif de VOR/DME (et non une fréquence) pour afficher la fréquence correspondante au poste VHF.NAV;

- un indicatif d'aérodrome et de QFU (par exemple LFPG 27 pour le QFU 27 à PARIS-CDG) pour afficher à la fois la fréquence ILS correspondante au poste ILS, et les paramètres de la piste (orientation exacte, longueur, pente du glide) au poste de commande du HUD pour l'élaboration de la piste synthétique.

Les postes de commande multiplexés permettront dans les années à venir à la fois de dégager les cockpits et de permettre un gain de poids notable.

6.2. LA TECHNIQUE "POUSSE-BOUTON"

Dans la perspective de l'allègement des cockpits, on ne peut passer sous silence le progrès spectaculaire apporté par la technique dite "pousse-bouton", qui a été particulièrement développée sur l'A.310.

Dans l'interface classique des avions commerciaux, une anomalie (ou une panne) est signalée par l'allumage d'un voyant lumineux. Pour identifier l'anomalie, l'équipage est conduit à relever les indications d'un ou plusieurs instruments, et à procéder pour faire disparaître l'anomalie à une action sur un actionneur électro-mécanique (bouton, basculeur, robinet ou autre).

Dans la technique "pousse-bouton", on remplace l'ensemble voyant/instrument/actionneur par un bouton-poussoir à éclairage intégré unique. Lorsque tout va bien, ce bouton poussoir est éteint. S'il vient à s'allumer, il signifie qu'une panne a été détectée, et qu'une pression sur lui-même produira les actions propres à remédier à la panne.

La procédure d'utilisation de ce système est donc la suivante:

- bouton-poussoir éteint = tout va bien

- bouton-poussoir allumé = on le presse, il s'éteint ou reste allumé d'une couleur conventionnelle signifiant "état dégradé" = la panne a été contrée. Pour plus de renseignements sur l'état résiduel du système concerné, il suffit de se reporter à un tube cathodique descriptif du système, comme présenté plus haut.

Une telle technique permet à la fois d'alléger le cockpit , et de diminuer la charge de travail et les risques d'erreur de l'équipage.

CONCLUSION

S'il est un domaine où il faut bien se garder de faire des prédictions, et de dépeindre l'avenir vingt ans, voire dix ou cinq à l'avance, c'est bien celui de l'aéronautique. Un contexte essentiellement changeant, dans le domaine technique mais aussi dans le domaine économique, rend aléatoire toute prévision à long, voire à moyen terme. Qui aurait pu prévoir, à l'apparition des Super-Constellation, que ces appareils seraient périmés avant même d'avoir volé, devant la poussée du transport à réaction ? Qui aurait pu prévoir, à l'apparition du Concorde, que le transport supersonique allait devoir attendre encore son heure quelques décennies ?

Malgré les incertitudes sur l'avenir, il n'est pas interdit de relever des tendances, et de prévoir avec certaines chances de succès que certaines seront irréversibles. Et dans le domaine de l'interface homme-machine, où la continuité des tendances semble plutôt être la règle, on peut imaginer que les points abordés plus haut, tubes cathodiques de planche de bord, systèmes de gestion du vol, collimateurs de pilotage et d'autres encore à peine dessinés constitueront les lignes de force du poste de pilotage des avions commerciaux de l'an 2000.

New Flight Deck Design in the Light of the Operational Capabilities

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Summary:

The classification of human errors (H.E.) by category and conditions of occurence is presented. According to the conditions and causes of H.E. occurence different means of preventing H.E.'s by ergonomic design, by manning, selection and training are considered. In flight deck design - besides conventional ergonomic design - two strategies are proposed:

(1) Provision of sufficient feedback to allow the pilot to detect and correct unintentional performance errors before they affect system performance.

(2) Introduction of "fail safe and fail ops. ergonomic design" by means of H.E. detection and correction functions designed into the man-machine interface intelligence.

The last generations of aircraft are compared regarding their task and system complexity. The increased complexity formed the need, and the advances in electronics provided the means for the new integrated flight deck, based on a cockpit data management system that includes "error monitoring capabilities". Finally the pilot's cockpit requirements of fire/flight control and post-stall-manoeuvring modes are discussed, requiring increased automation, information feedback loops and a "reconfiguration" of the pilot's task load.

Introduction

Increasing capabilities of to-day's fighter aircraft are based on fly-by-wire and digital electronics technologies, and on improved weapon and sensor concepts. We are designing poststall, coupled fuselage aiming, and other flight characteristics into the aircraft and are including advanced radar and electro-optical sensors and weapons into the weapon system. These modes and techniques require a flight deck enabling the pilot/aircrew to fly them, and - besides that - to handle all systems appropriately to effectively fulfill the mission.

The flight deck design task is:

To provide a work environment and situation equating the pilot's capabilities and needs.

This can only be done by a highly sophisticated Cockpit Data Management Concept, which provides:

- display functions selected by task and phase of mission
- display feedback on system and sensor functions selected
- control feedback to the pilot on control authority states in the new control modes
- task load monitoring
- fail safe/fail ops. monitoring of pilot's performance whereever applicable.

To achieve this design goal a certain knowledge on the nature of human errors and the conditions of their occurence is required.

Classification of Human Error (H.E.)

In the research concerning human reliability quite a number of classification schemes of H.E.s have been developed. All are based on one of the dimensions:

- category and/or type of H.E.
- cause conditions
- conditions of H.E. occurance

(1) Classification by category

Most classifications can be subsumed under the following categories:

a) Action error (substitution, reversal, adjustment etc.)

b) Sequence error (e.g. mix up of procedure steps)

c) Time error (out of phase or execution too slow etc.)

d) Omission error (forgetting of check, default action, step etc.)

e) Addition error (unintentional, unneccessary, unrequired, superflous action)

This line of thought is followed by Swain (1963), Meister (1964) and by others.

This classification deals with the error as such, not with the cause factors inducing the error. This approach of discriminating distinctly between a fact (e.g. incident or accident) and the cause or chain of cause factors behind that fact is known from accident research already (Seifert et al, 1980).

(2) Classification by cause conditions

Swain (1980) distinguishes between "Situation Conditioned Error" (SCE) and "Human conditioned Errors" (HCE). Embrey (1976) discriminates the "predisposing factors" as "Situational" or "Idiosynchratic".

As far as flight deck design is concerned, we have to consider how to prevent such situational conditions to manifest themselves in the cockpit design or in the weapon system operations. However before we discuss this task, we should highlight the third classification, which overlays the SCE/HCE classification, and concerns the conditions of H.E. occurance.

(3) Classification by conditions of occurence

Rigby (1969) distinguishes between systematic, random and sporadic H.E.s.

a) Systematic H.E.s

Both SCEs and HCEs can be conditioned systematically.

A typical example for a systematic SCE is: Substitution error between flaps and wheels lever. In the classic study by Fitts and Jones (1947) this error amounted to 31 % of the controls substitution errors. This is a systematic error induced by cockpit design. Even to-day there are aircraft of one manufacturer on the market, which are inducing this systematic error by their cockpit design.

Systematic SCEs can occur due to inappropriate selection methods and due to inappropriate training as well.

Systematic HCEs occur due to a bias in human performance caused by loss of fitness or motivation, by fatigue etc. That means a systematic out-of-tolerance degradation of human performance.

b) Random H.E.

Random H.E.s occur due to the randomness in the variation of human activity and performance. Even a fully qualified, trained, experienced, and practiced operator shows variations in performance. Random errors are explainable. They occur inevitably, due to overlooking, slipping, forgetting, erroneous retrieving, erroneous concluding etc.

They can not be excluded by conventional ergonomic optimization of the design, not by personnel selection and not by training and practice.

Random H.E.s are random only in terms of interindividual occurance. Intraindividually there is a non-random bias due to personality traits.

The "sporadic errors" as defined by Rigby (1969) are errors, that cannot be explained. They "just occur". Davis (1948) described them as "apparently unreasonable, and stupid mistakes", and named them "lapses".

With regard to flight deck design the sporadic errors or "lapses" can be treated as random errors. The only difference between the two is: random errors have causes which are more or less apparent, the lapses have not.

To summarize: A systematic H.E. is predictable, because identical situations (design failures etc.) result in the same type of error.

Random H.E.s are not predictable. They require a new approach within the system design task.

Treatment of H.E.s in flight deck design

Fig. 1 shows a general breakdown of H.E.s occurence and their dependency. In the lower part the responsibilities for H.E. prevention are named.

In conventional flight deck design systematic H.E.s are prevented by the application of ergonomic design procedures. With this conventional ergonomic design, we do our best to provide equipment functions, man-equipment interface, handling qualities and operating procedures, being in compliance with ergonomic requirements. This, however, is not enough in the light of the design of an "integrated cockpit", and the design of "new control modes" as specified for future fighter aircraft weapon systems.

Even if we are able - by ergonomic design - to preclude systematic H.E.s to occur, random H.E.s will happen to occur with a frequency of 1 in 100 to 1 in 1000 per unit measure (tasks, events, time cycles). Random H.E.s can not be prevented to occur. They happen - as has been stated above - due to normal variations in human performance. To reduce effects random H.E.s may have on system performance, a new strategy is required in flight deck design.

It is known that "humans" normally "operate" as closed-loop system, detecting and correcting their own errors (J.A. Adams, 1982). Consequently one design strategy is, that we provide feedback loops in the system or equipment functions, which support the human operator in detecting and correcting unintentional performance errors, before they can affect system performance. There is a second strategy, which can be applied in all cases, where human performance errors can be detected and evaluated by the system. For those H.E.s an error correction function can be designed into the system. This task is called "fail safe and fail ops. ergonomic design" (in Fig. 1). The goal of this second strategy is to achieve a certain tolerance of the technical system against human performance errors.

By the application of this design strategy we do not tolerate design failures, personnel selection or training failures, which lead to predictable systematic H.E.s. We only tolerate those human performance errors, which occur in spite of the compliance of the system and its interfaces with the defined human performance capabilities.

The design goal for new flight deck design is to provide mutual monitoring of man and the technical system. The human operator monitors and controls his system and his own performance, and in addition the technical system monitors the human operator performance whereever applicable.

Flight Deck Design Concepts

In the past new design concepts often were realized by simply adding new techniques or systems to those already installed. This resulted in extended flight capabilities and in greater mission flexibility. However, the pilot's workload increased as well.

Fig. 2 to 4 show block diagrams of aircraft of different generations. The blocks of new systems and those in which new functions (tasks, subsystems) were added are marked (●) with a dot. The increase of functions by number (displays, controls) is known from previous investigations (Ostgaard, 1981, and Lyons and Roe, 1980).

Fig. 5 and 6 show the follow-up from F-15, Tornado and F-18 to the 1990 fighter aircraft. It can be seen that the "cockpit integration" leads to a reduction of displays ranging from 40 % to 50 %. The controls are reduced only by 20 % to 25 %. This supports the conclusion drawn in earlier investigations (Seifert, 1981, and AFFDL, 1977) that multifunction controls are even more important and intricate in terms of success of the "digital cockpit" than multifunction displays.

For the future fighter aircraft this additive concept is no longer acceptable. The F-18 represents the first generation of aircraft with a flight deck, to which the term "integrated cockpit" applies. Yet even the F-18 does not mark the end of flight deck design and aircraft capabilities. The future fighter aircraft has the following features:

- Flight deck with high functional flexibility, based on the installation of a "cockpit data management system" (CDMS), as presented in Fig. 7 and 8.
- New performance modes "coupled fuselage aiming", "poststall manoeuvrability" and possibly "direct lift control", "drag modulation", or other manoeuvre enhancement modes.
- Extended digital avionic and digital flight control systems using Mil-Bus data highways.

The resulting tasks for the flight deck design are:

a. Monitoring of task load and information flow in the cockpit, to limit pilot's workload in all flight and mission phases.

b. Provide information feedback to the pilot where appropriate to enable him to monitor the validity of this performance.

c. Provide appropriate information to the pilot, in order to avoid undue reactions (even spatial disorientation) by strange motion cues during new extended mode manoeuvres.

d. By allowing for ergonomic design, decrease the probability of systematic H.E.s (pilot errors).

e. Increase the ability of the system to tolerate and compensate H.E.s (pilot errors) where applicable.

Only if these tasks are accomplished by the new flight deck design, the pilot will be enabled to fully utilise the extended system and performance capabilities of the future fighter aircraft.

Cockpit Data Management System

A first concept for handling the data management in an integrated cockpit was presented in an earlier AGARD conference (Seifert, Denkscherz, 1977). This concept included:

a. Moding by flight or mission phase

b. Integrated display and display information flow management

c. Pilot's authority to operate individual systems within or outside the mission mode selected.

Meanwhile the concept was realized. However, with the realization new ideas are formed to upgrade this concept by an error management feature for pilot command and data input functions. The block diagram of the cockpit data management concept is presented in Fig. 8.

From Fig. 8 it can be seen, that the new flight deck design is based mainly on multifunction displays and controls, and on the MIL-BUS application. With increasing confidence in and proof of the reliability of MIL-BUS systems, the "direct wired" data flow between aircraft systems and cockpit will decrease further. Even the backup instruments and warnings may be wired through the MIL-BUS by then.

From the multifunction concept follows, that the ergonomic design tasks are more and more encompassing both, the hardware and the software specification. In this context the terms "error management" and "error tolerant" in relation to human error emanated and became meaningful. Recently "error management requirement of such software dependent man-machine interface" were defined by S.L. Smith (1981).

Smith defined 16 basic error management requirements. If you closely look at them, they can be categorized as follows:

(1) H.E. monitoring with automatic correction and display feedback, or with feedback for correction by operator.

(2) Protection of system (store, stack) against H.E.

(3) Allowance for the operator to correct self-detected errors.

(4) Interface functions logic layout requirements

We transformed this information to allow their incorparation into the requirements for new flight deck design:

(1) H.E. monitoring with automatic correction and feedback, or with feedback for correction by the pilot.
This capability should be installed in conjunction with command entries and data entries regarding certain typical H.E.s:

- o Action errors
  Entries which are logically inappropriate, unidentifiable (e.g.voice), or ambiguous

- o Sequence errors
  Invalid step within an entry sequence, e.g. in navigation data entry or system functions command-sequence.

- o Omission errors
  Default command or data entry.

- o Addition and Time errors (according to H.E. classification) are obviously subsumed under the above action errors in Smith's repertiore. They have to be considered here, e.g. a function command that is out of phase within the system control in a given mission phase.

(2) Protection of system against H.E.

Certain data manipulation actions should be made safe against human decision and command error. Smith (1981) recommends an operator self-check.

In critical situations concerning integrity of program and data store the system requests the operator to "confirm" a command entry before it is executed by the system.

This is recommended to be done in case

- o an extensive change (e.g. deletion or mission planning alteration) of stored data is commanded
- o of log out command is given (e.g. out of a check procedure) before other pending transactions were performed.

(3) Allowance for the operator to correct self-detected own errors.

This is based on the "closed loop performance" characteristic of man, as detailed by Adams (1982), which was mentioned earlier.

The system shall allow for:

- o Immediate correction of entry i.e. while the entry is still in mind and the source data still at hand.
- o Easy return to previous steps for correction of data or sequence error or introduction of desired change, e.g. extended command entry.

(4) Interface layout

This category deals with the availability, consistency and flexibility of use of entry functions:

- o Layout and consistency of ENTER function
- o Feasibility and layout of a CONFIRM function
- o Flexibility of data change and function command entries within a master mode.
- o Data entry consistency over the displays and keyboards in the cockpit.

All these design tasks are typical software ergonomic tasks. It is not "error management" in terms of H.E. monitoring through the cockpit data management system. According to the classification of H.E.s as shown in Fig. 1, this interface layout shall be such, that "system induced H.E.s" are prevented to occur.

To summarize the error monitoring tasks within the CDMS, this system shall in this priority sequence:

- o detect and correct pilot entry errors
- o detect entry errors, and display message for the pilot to take corrective measures
- o detect potential loss of stored data due to pilot action, and ask pilot for advice
- o by information feedback on pilot actions make allowance for pilot's capability to correct self-detected own errors.

Advanced manoeuvrability modes

Since the "advanced fighter technology integration" (AFTI) program has been initiated (Martin, 1973, Wetmore, 1976), a number of projects evolved, aiming at higher manoeuvrability and more sophisticated weapon system capabilities. There are the "F-15 Integrated Fire Flight Control" (IFFC) program, the Advanced Manoeuvring Attack System (AMAS), and similar European programs.

Not all of the programs include a capability for a six-degree-of-freedom control. However, any enhanced manoeuvrability has to rely not only on an advanced display/control concept, but also on a higher degree of automation.

This increased automation in trajectory and attitude control requires new information feedback loops, to keep the pilot informed about system states, processes, modes and about the authorithy range and limits, in which he is operating. This has been recommended already in the report of the "Committee on Automation in Combat Aircraft" (Lowndes, 1982).

In to-day's aircraft we have installed emergency-prevention devices: Examples are:

- o angle of attack and g-force limiters (F-16)
- o stall indicators (F-111)
- o stall prevention and incidence limiting system (TORNADO)

With these devices installed, and with the application of the fly-by-wire technology for flight and propulsion control, we have the means available, to give the pilot all the information he needs, and to relieve him from tasks, he no longer is able to accomplish.

(1) Fire/flight control modes

In this mode the following control functions are automated:

- o attitude and trajectory control after lock-on
- o target detection, acquisition and identification
- o target priorization.
- o weapon launch (missiles)

By this automation greater system effectiveness is achieved. The pilot's workload does not decrease. However, the automation of the continuous control functions allows to "reconfigure" the workload of the pilot. He is enabled to keep his mind free for tactical and attack monitoring tasks. The manual functions left are:

- o mode selection
- o propulsion control (if not automated)
- o sensor system control
- o weapon launch (gun, rockets)
- o override of automatic control and of target priorization.

Besides these manual functions, there is a definite requirement for information feedback to the pilot regarding the automatically controlled functions. These feedback display requirements include:

- o attitude and trajectory (flight director) information
- o target and sensor information
- o authority limits of fuselage aiming
- o shoot range.

In case propulsion control is automated as well, additional feedback information is not likely to be required.

(2) Poststall mode

This mode requires automated

- o attitude control
- o g-load and angle of attack limitation.

In this mode the information feedback to the pilot becomes extremely critical. The reason therefore is the extreme departure of the trajectory from the attitude of the aircraft. This results in an excessive incompatibility of the visual and the proprioceptive sensational information of the pilot. To prevent pilot disorientation, this mode requires:

- o proprioceptive feedback about going into and leaving the poststall mode
- o display of information on the relation between attitude and trajectory
- o display of information regarding the state of the aircraft within the mode's autority limits.

Concluding Remarks

New sensor, flight control and attack functions require cockpit integration and a considerable amount of automation. Therewith the pilot's task is reconfigured and the workload is kept on an acceptable level. To enable the pilot to fully deploy his skills in such a system, further cockpit system functions are required, such as: pilot error management, and display of specific feedback information in conjunction with data handling, and with the new flight control and attack modes. -These cockpit system function requirements are discussed.

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Human Errors				
Main cause	Situation conditioned errors			Human conditioned errors
	System conditions	Work conditions	Personnel conditions	
Cause categories	System design • Work system • Work position Equipment design • Hardware • Software • Tools • Manuals Task design • Sequences • Task alternation • Time consumption	Work environment • Weather • Threats • Mission task load Work organization • One man work • Teamwork • Command, control communication (CCC)	Personnel requirements • Selection • Manning • Instruction • Training • Practice	Intraindividual performance variations • Perception • Processing • Storage, retrieval • Output action • Arousal level
Prevention by	Ergonomic design	Specification of 1) Mission requirements 2) CCC requirements 3) Effectiveness and reconfiguration requirements	1) QQPR*) specification 2) Selection 3) Training	"Fail safe, fail ops." ergonomic design, Error management system design

*) Qualitative and quantitative personnel requirements

Fig. 1 Human error classification

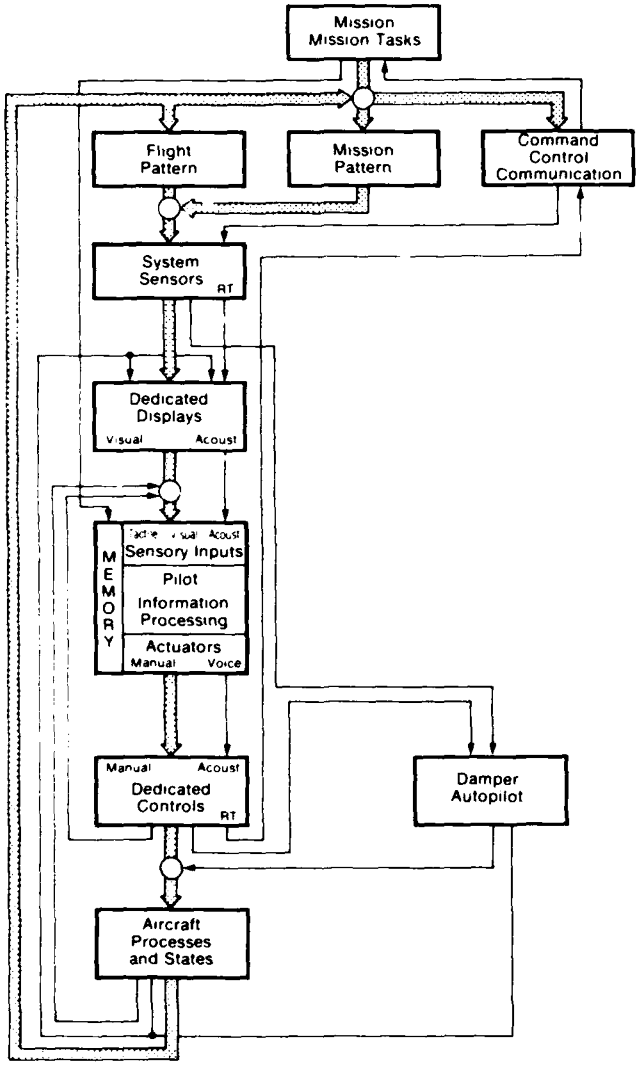

Fig. 2 Augmented System (F-104)

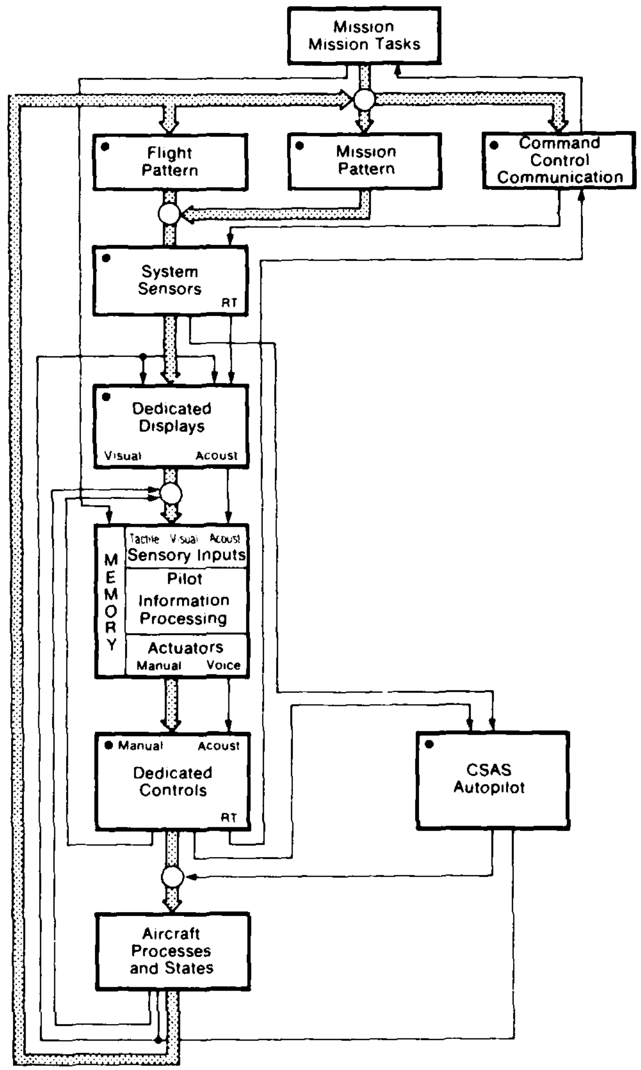

Fig. 3 Augmented system (F-4)

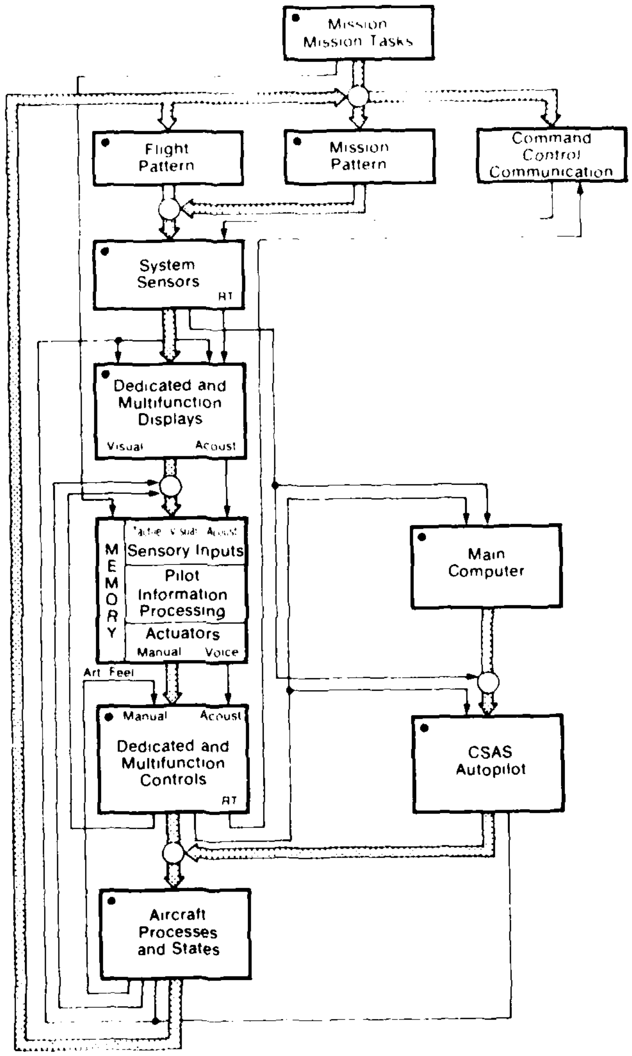

Fig. 4 Highly augmented system (TORNADO)

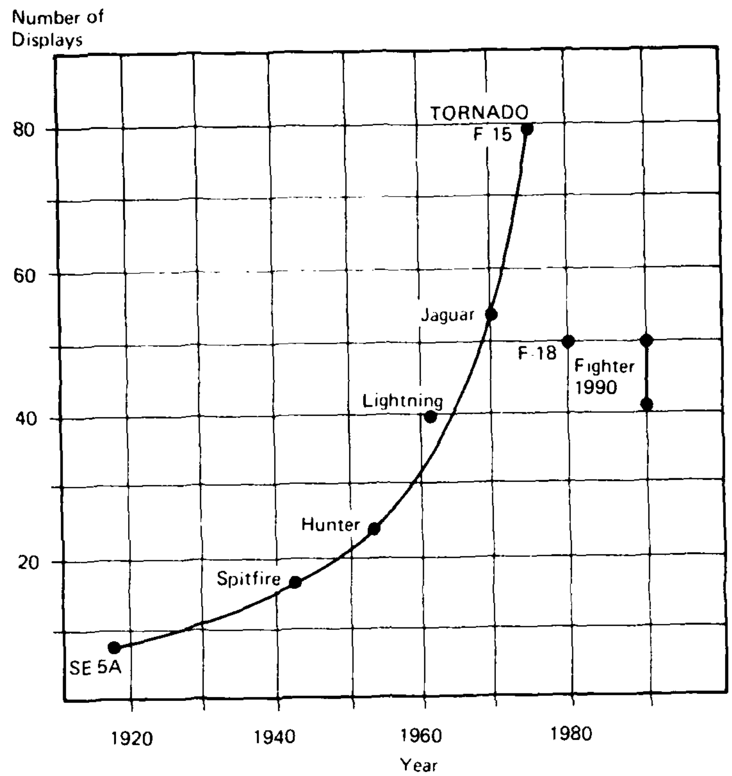

Fig. 5 Number of displays in the pilot's cockpit over the generations of fighter aircraft

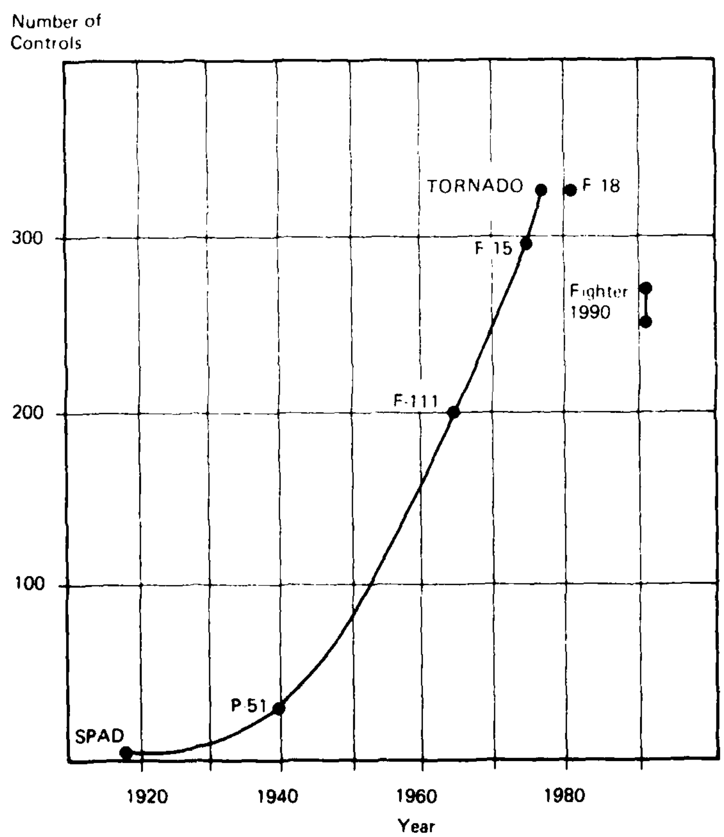

Fig. 6 Number of controls in the pilot's cockpit over the generations of fighter aircraft

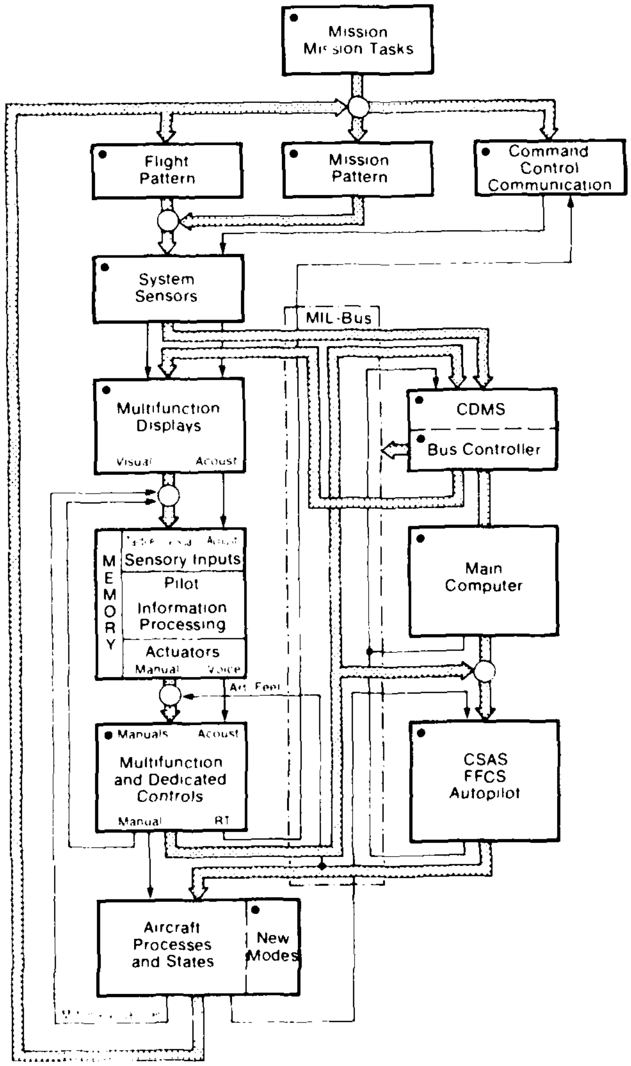

Fig. 7 Superaugmented system ("6-degrees of freedom")

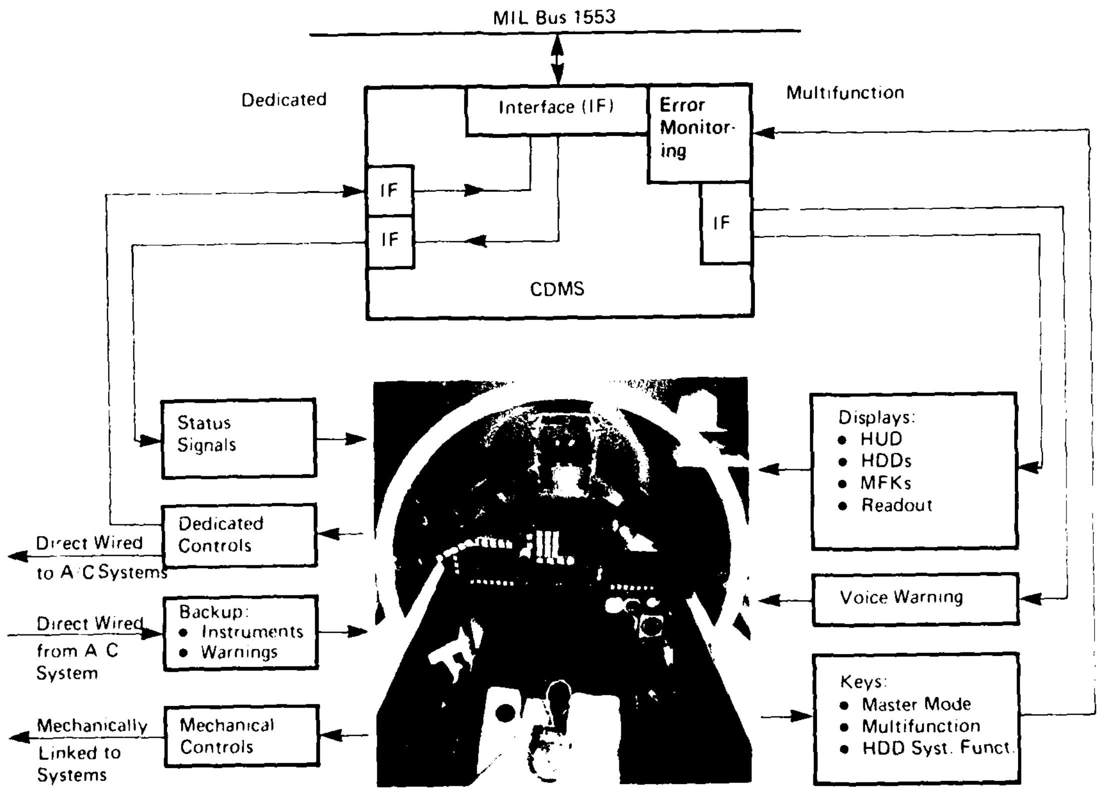

Fig. 8 Cockpit Data Management System

AD P002716

MODERN FLIGHT INSTRUMENT DISPLAYS AS A MAJOR MILITARY AVIATION FLIGHT SAFETY WEAKNESS

by

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SUMMARY

Consideration of the major causes of flying accidents over which the airframe and engine manufacturers can exert a powerful influence shows the following list :

1. Structural Failure
2. Engine Failure
3. Flying Control Failure
4. Instrument Failure
5. Pilot Error.

With the first three of these causes - Structural, Engine and Flying Control failures - while mistakes do occur, the manufacturers have a reasonable record, there is no evidence of complacency, and in addition there is a large well established, government controlled, national bureaucracy offering valuable checks and advice on testing and airworthiness certification. Pilot error in different, but appropriate ways, also attracts much effort aimed at its reduction. Most importantly, so far as the purposes of this paper are concerned, the accident trends related to the first three causes, as well as those due to pilot error, do not appear to have changed fundamentally during the last decade. The same cannot be said of instrument display related accidents.

Since the advent of Head Up and computed displays in general, and the operator's real need to expand the non-visual manoeuvre envelope, there has been a marked increase in display related accidents/incidents in both operational and development flying.

This note suggests that attempts at curing the problem have been based on a false assumption that has ignored the reality of the piloting task in modern high performance jet aircraft. Proposals are offered to improve the situation by both engineering and organisational changes.

1. THE SITUATION IN THE 1950's

Service aircraft used traditional mechanical and electromechanical direct reading individual instruments with well understood limitations and failure modes. Some accidents occurred due to misreading particularly of altimeters. The operators asked for a greater manoeuvre envelope while flying on instruments as well as improved display forms to reduce pilot error problems.

2. THE SITUATION IN THE 1960's

Head down OR 946 instrument displays started to be used extensively. These were easier to read and with the use of Master Reference Gyros gave an improvement in manoeuvre capability. They brought some new failure problems and aircraft fitted with them were at times restricted to visual conditions while improvements were incorporated. However, the overall situation was not fundamentally changed from that existing in the 1950's.

3. THE SITUATION IN THE 1970's

The first aircraft to be fitted with Head Up Displays as a primary instrument flying reference entered service in 1969. The head down instruments were limited in nature and number and were conceived as get you home standbys to be used in the event of HUD failure.

Head Up Displays as fitted to the aircraft in the early 1970's showed quite fundamental differences from previous flight instrument displays. These were :

(a) The symbology was very easy to use and for any given manoeuvre gave a large reduction in instrument flying pilot workload.

(b) The freedom from roll and pitch limitations given by the inertial platform attitude reference sources encouraged pilots to manoeuvre, without visual references, more violently than previously.

(c) The data displayed could be wrong without this being in the least apparent to the pilot. (For instance because the display wave form generator correctly processed an input that was itself incorrect.) Such correctly WRITTEN information looked totally valid to the pilot.

(d) The displays could be quite compelling.

(e) The reliability of the overall display system proved to be very low. It was not unusual for individual pilots to experience some sort of failure each month.

4. REACTIONS TO THE SITUATION IN THE 1970's

The reaction of all interested parties to this unprecedented lack of reliability was to stop considering the HUD as the primary flight instrument display and to call for the pilot to cross refer between the HUD and the standby instruments.

In the writer's view this is where we went wrong.

This solution, which called for the pilot to cross check various displays to establish which were serviceable, ignored several important facts :

(a) In conditions of high workload the pilot will abandon all but essential tasks and revert to the easiest display from which to get the information he needs. Unquestionably this will be the HUD symbology since it includes nice easy to use items such as inertial vertical speed, climb dive angle, velocity vector and so forth. THIS LEADS AUTOMATICALLY TO THE LEAST RELIABLE INFORMATION BEING RELIED UPON AT ALL TIMES OF CRITICAL PILOT WORKLOAD.

(b) At times the head down instruments can topple, as they tend to have restrictive manoeuvre limits, leading to confusion if cross reference is used, and the chance that the pilot will believe the more reliable head down displays which are this time wrong because the original manoeuvre was one that only the HUD system could cope with.

(c) Cross reference between several displays makes instrument flying HARDER NOT EASIER. It can force good old fashioned mistakes associated with scan failures between various parameters BECAUSE THE PILOT WAS SCANNING FOR CROSS REFERENCE PURPOSES between head up and head down examples of the same parameter.

The writer firmly believes that it is unrealistic to rely on the pilot cross checking various read outs of nominally the same information in order to establish serviceability. One recent classic example of where a 3 man crew equipped with 3 attitude references did not notice that the handling pilot was following an incorrect one, was the Air India 747 that rolled inverted and crashed, shortly after a night take-off out of Bombay.

5. GENERAL PROPOSALS TO IMPROVE THE SITUATION

(a) Engineering

Ideally the flight data should be multiplexed, the validity of data should be decided automatically in the back of the aircraft and the pilot sh[illegible]ld be presented with this reliable data through both head up and head down displays. There is no justification for treating cockpit display data as somehow less important than, say, wing spar strength or the signal to the tailplane jack in auto terrain following or auto land cases.

Since the ideal solution will require much time and cost to [illegible]corporate, an interim solution offering some level of improvement (over the present case of hoping the pilot spots which instrument or symbol is out of step with the rest) is to have duplex systems with a monitor lane that warns the pilot through an audio warning that a display disagreement exists. Leaving him to use his skills to diagnose which display is at fault.

In this connection the Royal Aircraft Establishment Farnborough have started work on how to tell the pilot visually that a flight instrument system failure has happened. Even this task has been harder than expected, for instance, writing a large cross on the display is certainly attention getting, but in some flight trials pilots have momentarily tried to use the failure indicating cross as a reference. Merely removing the invalid information from the display has been shown to be too weak. In this case too much time can pass before the pilot realises it has been removed. Time which had he been ALERTED, could have been used to avoid a subsequent critical situation.

(b) Organisational

Because modern displays can use several discrete black boxes in series to produce the end result, NO ONE MANUFACTURER/DESIGNER MAY BE RESPONSIBLE FOR MORE THAN ONE ELEMENT. Because each component manufacturer is in turn only interested in showing that his element meets the specification requirements placed on him, the full interactions of the total system may not be appreciated, especially the knock on effect of any changes that may be made in isolation in one box. By its nature, this sort of problem is less likely to occur in Structure, Engine and Flying Control disciplines (But watch out for fly-by wire).

The effects of limited responsibility for one link in the chain can only be mitigated by instituting an overall system co-ordination function that must itself exist in all of the interested organisations - manufacturers, airworthiness authority and customer - if the full checks and balances that we have come to rely on for Structures, Engines and Controls are also to be available for Flight Instruments.

6. CONCLUDING REMARKS

At present it is a sad fact that with some modern aircraft only the President of a Board of Inquiry has the motivation combined with the authority, to investigate the total system operation so far as displays are concerned.

We must develop the deep infrastructure for displays that exists in other aircraft design disciplines. From the standpoint of aircraft survival, instrument displays merit at least the level of emphasis accorded to structural, engine or control factors.

No relevant authority, research establishment, licencing organisation, manufacturer or operator should be content with the present situation. It is no longer responsible to keep telling pilots to cross check better, and to imply that if they crash through a displays problem, then they are in some way not professional enough in their approach to flying.

PHYSIOLOGICAL AND PSYCHOLOGICAL ASPECTS OF THE PILOTING OF MODERN HIGH-PERFORMANCE COMBAT AIRCRAFT

by

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Since 1978 the Royal Netherlands Air Force has gained a lot of experience in handling a totally new aircraft, the F-16, which differs in many aspects from its predecessors. Not only the pilots, but the technicians too, had to adapt themselves to the new aircraft and they learned how to handle entirely new systems.
The F-16 is a modern high performance fighter, like the F-14 and the F-15, and has caused problems for the pilots in comparison with former types, the F-104 and the NF-5. The extreme mobility of the aircraft produces not only a high gravity factor as such, but also a rapid G onset rate, by which a number of gravity depending physiological problems have arisen in a very obvious way. Furthermore as a result of many technological innovations applied, i.e. aircraft's electronics, the pilot's task has also changed considerably, in particular with respect to the amount of information to be processed. The question arises of wether new approaches should be introduced into the selection, the physiological standards, the training, the supervision and the coaching of the operational pilots.

PHYSIOLOGICAL FACTORS

Almost all problems on the physiological, the medical and the pathological field have something to do with gravity.
Any human is able to withstand in an untrained status and without aids an acceleration of 3,5 - 4,5 G in the head to foot direction (+Gz). Acceleration is defined by Webster's dictionary as "a change of velocity, or the rate of such change, either as regards rate or direction or both". The force of gravity, or G, is it's unit of measurement.
It would probably be opportune here to explain the different types of G vectors:
+ Gz is foot to head.
Acceleration or 'eyeballs down': -Gz is head to foot.
Acceleration or 'eyeballs up': + Gx is back to front.
Acceleration or 'eyeballs in': -Gx is front to back.
Acceleration or 'eyeballs out' and ± Gy are acceleration forces applied from the left or the right and consecuently 'eyeballs 'left' or 'right'.
When evaluating G-tolerance levels, one must consider the following parameters: direction of G vectors, duration of G, magnitude of G, time at peak G, rate of G onset and decay, endpoint of tolerance selected, environmental conditions present, anti-G mechanisms used, body positioning and the motivation, conditioning and training of the pilots.
In flying the F-16 the + Gz and - Gz is of interest. + Gz causes fatigue, perspiration, cough (especially if oxygen is being used), a feeling of warmth, calf pain, Peripheral light loss, Central light loss, loss of hearing and unconsciousness, occasionally with convulsions, in that order. As mentioned before humans can easily withstand acceleration forces up to + 4,5 Gz, still accelerating the pilots develop the afore-mentioned problems because the pulse pressure (systolic minus diastolic pressure) is no longer sufficiant to force blood to the head, this is due to the increase in weight caused by high G load. The eye is the first organ to deteriorate because the normal eye pressure of almost 20 mm Hg is increased, therefore it might be better to warn pilots in cases of emergency by acoustical signals and not primarily by caution lights.
There are various methods for modifying G-tolerance.
Among these are varying the G onset rates, water immersion, positive pressure breathing, the use of hormones and drugs, use of the M1 and L1 maneuver, use of the anti-G-suit and varying the seatback angels. In spite of the above, there is no practical means of counteracting the effects of negative acceleration, which causes "red vision", headache, mental confusion, muscular uncoordination and unconsciousness. Both the M1 maneuver - forced expiration with the glottis partially closed, while tensing leg and abdominal muscles - and the anti-G-suit can add about 1,5 to 2,0 G to one's tolerance. The M1-maneuver, however, is quite fatiguing and cannot be performed for long periods of time. Special physical conditioning programs based on weight- training for the upper part of the body may facilitate doing the M1-maneuver. Epperson found a 75% increase in G-tolerance after a 12 weeks muscle training program.
It seems as if some preinflation of the anti-G-suit may also increase one's tolerance. Leverett and Shubrooks have reported that positive pressure breathing, especially in the more Gx configurations, provides an increased G tolerance equal to that attained with the M1 maneuver with less fatique. Positive pressure breathing is only of value when an anti-G-suit is used and it can cause atelectasis of the lung, especially when 100% oxygen has been inspired.
Positioning the subject from a + Gz position to a more + Gx position, was repeatedly found to increase tolerance. Maximum protection is given in a up to 85° backward tilted seat, but this causes some discomfort for the chest. The F-16 tilt-back seat has only a 30° profile and will not offer a significant increase in tolerance values in horizontal flight figures, but in Air Combat Maneuvering one has to add the angle of attack of at least 15°, and the 45° tilt-back does offer some benefit. The PALE (Pelvis and legs elevating) G protective aircrew seat, however, gives excellent protection without restrictions.

Furthermore, hypothermia increases G tolerance; dehydration, dietary restriction of sodium, heat stress, prolonged bedrest and sunburn decrease G tolerance.

The high + Gz onset rate of the high performance fighter aircraft does not allow the body to accommodate in the proper way, therefore G-tolerance will be decreased by comparison with a slower onset rate. High sustained G can cause some irregularities in heart function.

Froehlicher found a gravitation depending Wolf Parkinson White preexcitation conduction pattern; Whinnery described coincident loss of consciousness and ventricular tachycardia during + G stress due to an imbalance between sympathetic and parasympathetic tone. Especially sportsmen or pilots in very good physical condition, caused by endurance sports, can show an increased vagal tone, which seems to be the etiology of the heart rythmn disturbances. Vettes and Vieillefond found that heart rate increase, always observed during acceleration, is a function of acceleration intensity. Tachycardia is an immediate reflex response originating in the carotid and aortic baroreceptors. The cardiac is reduced by 32% at + 4Gz and by 34% at + 5Gz in spite of tachycardia.
Some heart syndromes will produce arrythmias e.g. the mid systolic click - late systolic murmor syndrome, associated with dysfunction of the mitral valve, can give ventricular tachycardia or ventricular bigeminy. Therefore mitral valve prolapse makes the pilot unfit for high performance aircraft. Whinnery found an A-V dissociation during that +Gz exposure poses no significant risk for cardiac damage in humans and it will be up to the flight surgeons to select only those pilots who can withstand +Gz stress and who are completely healthy. Completely, because pilots who suffer with relatively mild or even subclinical deseases may become suddenly incapacitated as a result of high sustained +Gz. Mills and Marks have investigated the effect of acceleration stress upon the concentration of various hormones in peripheral blood. No significant changes were seen in the levels of GH, PRL, TSH, LH and FSH. Changes in serum cortisol were significant with peak values occuring 20 minutes after acceleration stress. The absence of significant changes in GH or PRL is not in accord with current evidence that they are "stress hormones".

Another problem. The retro-hyperflexion luxation of the cervical spine, has already been found after ejections. But sudden high +Gz loads in the new fighter aircraft can produce the same results. This happens when the torso is relatively fixed because of its attachment to the seat and the head is in a relatively fixed position because of the helmet which will be pressed against the headrest, then the neck is free to move. Furthermore, the antero - latero - hyperflexion, which causes extreme muscular pain is more common, because the +Gz load is not only working on the head but on the helmet too. If the neck muscles are not strong enough to withstand the forces, a luxation of the cervical spinal column or sprain of the cervical muscles will be caused.
A new type of helmet is now being introduced in the RNAF. The weight of the helmet including the visors, the oxygen mask etc., will be reduced [illegible].
Leverett described cutaneous petechial hemorrhages in the form of minute points or larger spots, deeply coloured, but painless disappearing spontaneously in a few days. Such injuries mostly occur in the lower part of the body and the underarms.
It is curious that petechial hemorrhages seem to develop in the first high performance missions. After some experience in flying this type of aircraft petechial hemorrhages are seldom seen. It is not surprising that due to the physiological problems in high performance aircraft flying the flight physical standards had to be renewed. Besides an extensive initial examination in the Netherlands a total spine X-ray is made in order to demonstrate changes in the long run, an EEG ampex to demonstrate changes in brainfunction. Furthermore there is an intention to use the human centrifuge in Soesterberg, Holland for training and selection purposes. After renovation, this centrifuge will be capable of being flown by the pilot using target tracking on a monitor inside the gondola.
Other countries have shown interest in using this centrifuge as well. Since 1980 van den Biggelaar and myself have fitted the flight physical standards to the new aircraft, they are almost equal to the standards for high performance aircraft pilots being proposed by Hickmann and others to the USAF.
The F-16 using countries have been united in the "F-16 Medical Working Group", under chairmanship of my medical inspector Air Commodore Maat, M.D.F.S., semiannual medical problems are being discussed. A special training program based on the principles of weight lifting has been developed in order to strength the muscles which are used in the M1 maneuver.

PSYCHOLOGICAL FACTORS

In recent years, increasing societal demands for safety, dependality, economy, effectivness and reduced energy consumption have increased the complexity of military flying operations, thus magnifying the pressures for good pilot judgment.
Furthermore technological advances that have eased much of the pilot's burden for precise aircraft control and weapon delivery have not eased the pilot's decision-making workload. In many cases these advances have only created demands for higher levels of skills, knowledge and judgment for which few pilots have been trained. The effects are distinctively noticeable in two areas, namely:

a. with respect to the psychomotor skills. Purely manual flying is not possible anymore. More and more, the pilot only gives input signals to be tested by computers for their applicability dependent on variables which are measured beyond the control of the pilot. Overcorrections and a number of incorrect inputs are not allowed or are corrected. The much vaunted "pilot's feel" has become less important. This way of flying, controlled for instance in the F-16 by a flight control computer, can cause vertigo. Vertigo is a sensation of spinning which may be caused when information from the eyes, body orientation and other sensations are not giving the same information.

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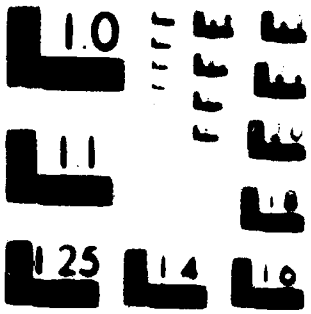
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The [illegible]

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[illegible] and any interaction between the two. [illegible] at a different height speed [illegible] At each height speed [illegible] there was a period of flight with and without data entry so that the effect of data entry on flying performance could be measured directly.

The order in which the systems were flown was randomized [illegible] learning effects. The latitudes and longitudes entered were [illegible] The [illegible] itself was kept [illegible]

Each pilot flew 4 missions using each of the [illegible] systems. Additionally, [illegible] using an Eye Mark camera to study the pilot's operating strategy as an aid to explaining the results from the main experiment.

6. Experimental Procedure

Sufficient briefing and training time was allowed for the subjects to become familiar with each system in turn. The first part of the training was confined to operating the particular system alone. This was then extended to include the flying task as well so that the practice and working environments were identical apart from the subject being aware that he was training.

Training for the [illegible] system [illegible] particular [illegible] were [illegible] by the end of the training [illegible] equipment was considered to have [illegible] experiments [illegible] not arrived at until this was the case.

The experiment considered of [illegible] data entry [illegible] over the [illegible] and written down by the pilot [illegible] entering [illegible] the system. [illegible] the experiment was [illegible] of writing of the data [illegible] waypoint numbers and contents were prepared [illegible] was then fitted to the pilot's kneepad [illegible] the experiment [illegible] the results [illegible] waypoint [illegible] There were [illegible] possible waypoints [illegible] particular [illegible] were minimised.

[illegible]

[illegible]

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[illegible]

[illegible]

[illegible]

[illegible] the eye mark camera [illegible] measure the lengths of time that the pilot spent head down [illegible] navigation data. The term "head-down" was interpreted as [illegible] display. Thus looking at the keyboard and the display [illegible] was regarded as a head down glance.

Figure [illegible] cumulative frequency [illegible] head down times taken from [illegible] data entry sequences for each system. These sequences were shared equally between [illegible] subjects and between the [illegible] speed height combinations. However there was no significant difference at the [illegible] level between the times for the [illegible] flight conditions. The head-down times for the touch sensing display and keyboard were not significantly different at the [illegible] level. However the head-down times for the [illegible] system were significantly shorter than those for the other [illegible] systems at the [illegible] level.

[illegible]

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The error rates [illegible] were very large throughout the experiment and were greater for the higher speed [illegible] than the lower speed [illegible]. This was despite a [illegible] practice and training programme which was [illegible] for each pilot until he considered he was a [illegible] [illegible] recognition during practice sorties. The errors were attributed to changes in the speech [illegible] of the pilots due to increased stress between the practice and trial sorties and between the low and high speed sorties. Some pilots were better at using the equipment than others, individual error rates varying between [illegible] and [illegible].

The [illegible] data [illegible] Table [illegible] data entry [illegible] adversely affected [illegible] systems.

[illegible]

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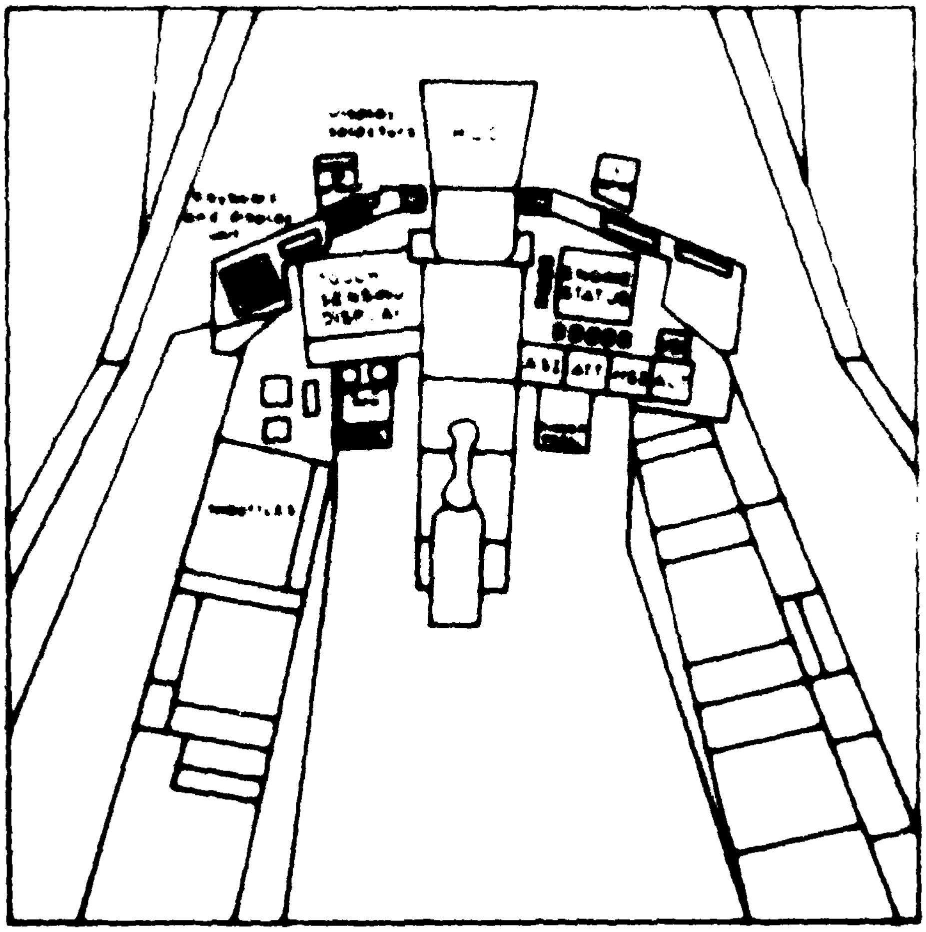

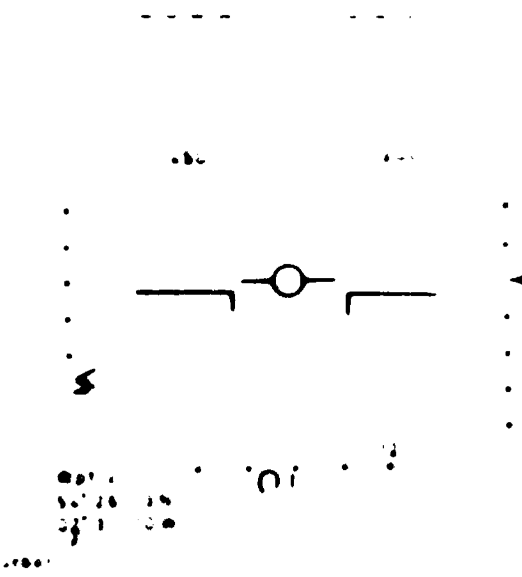
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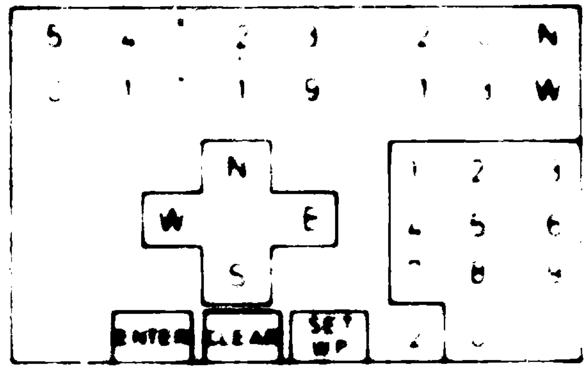
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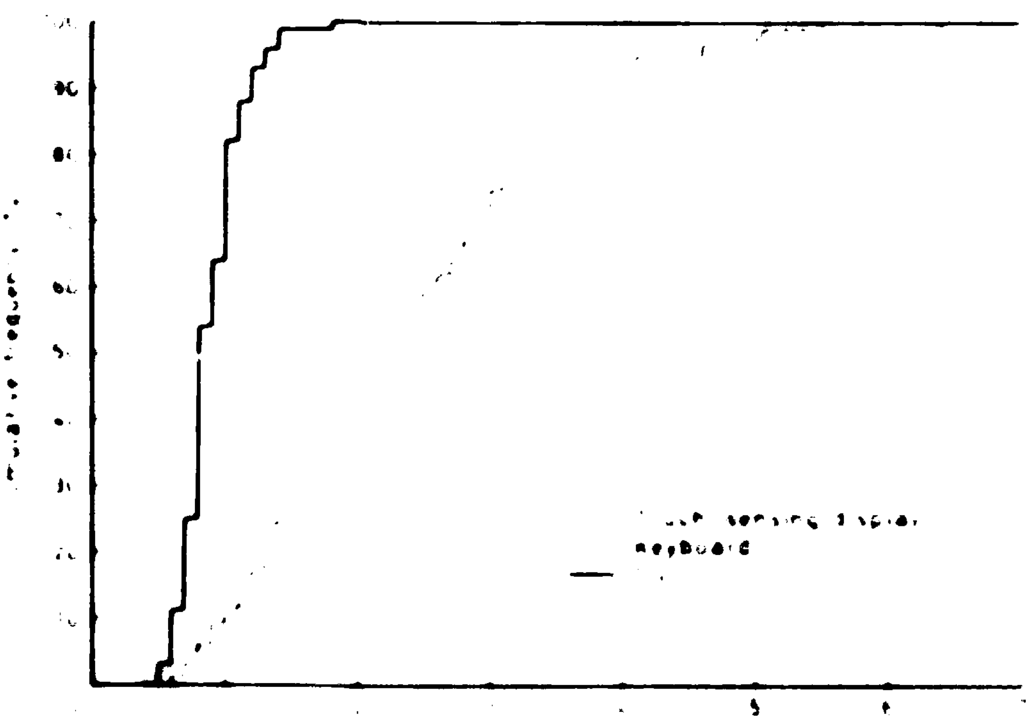
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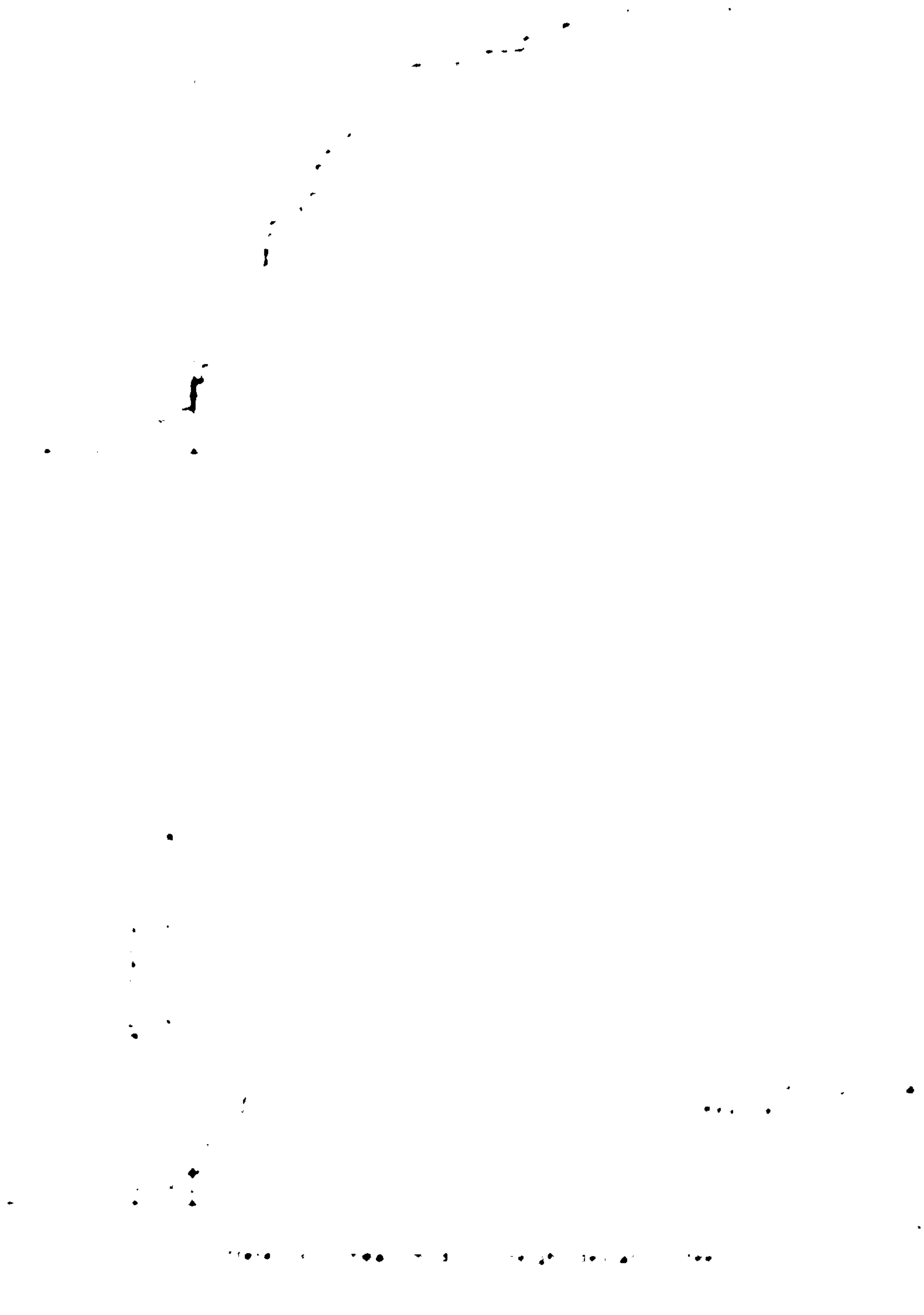
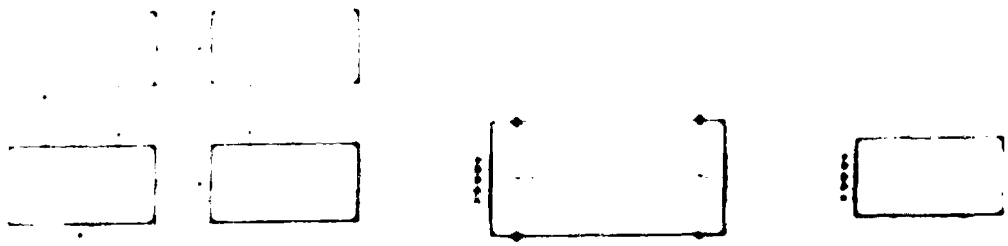
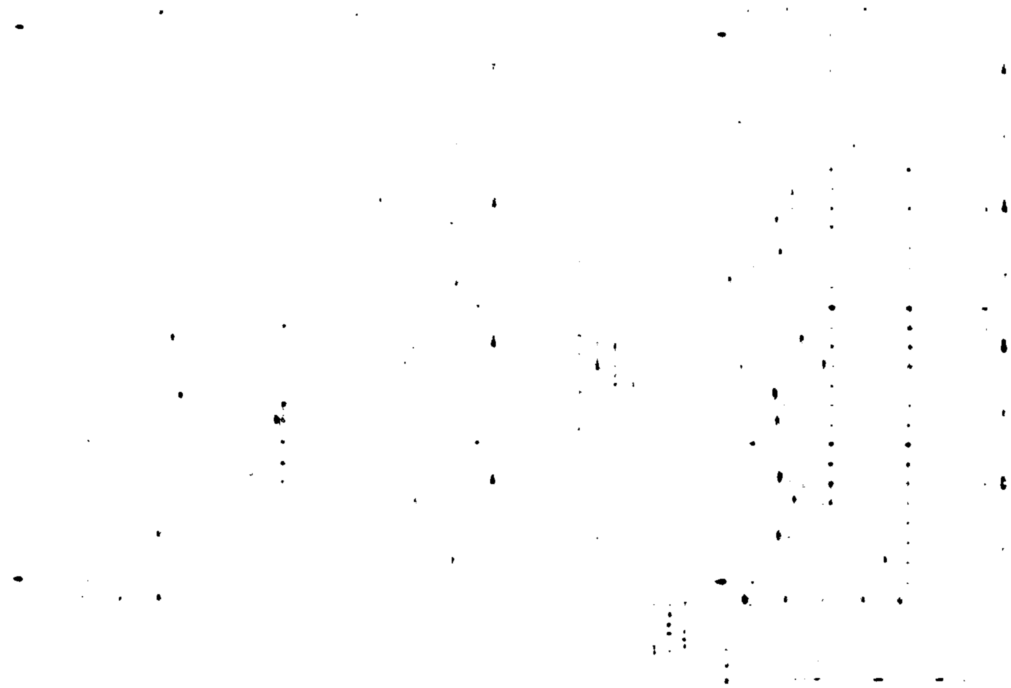
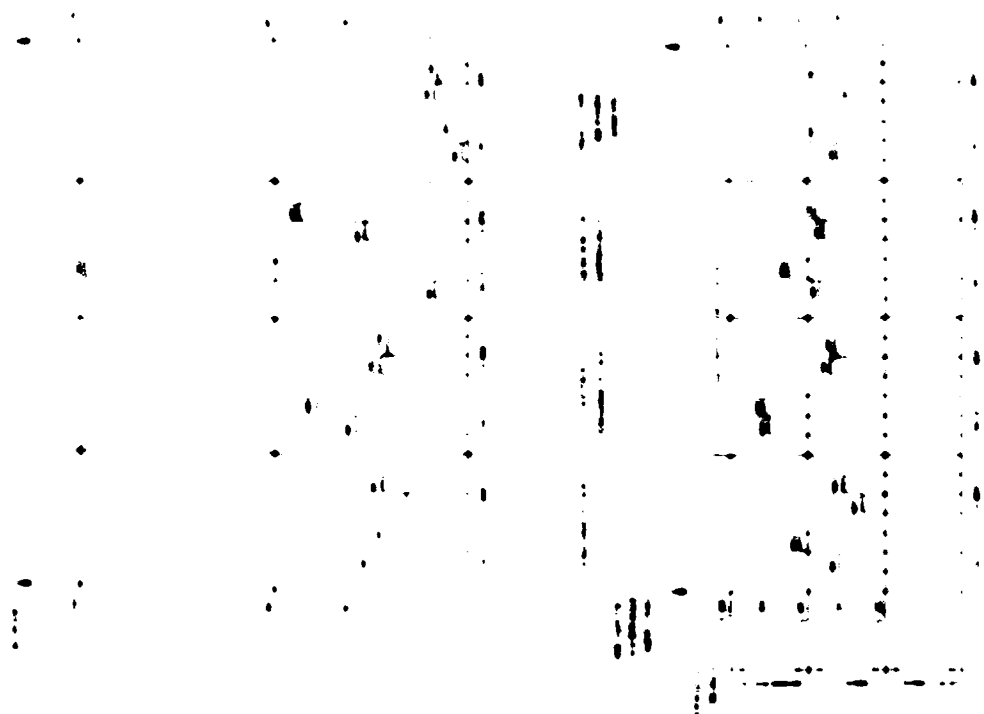
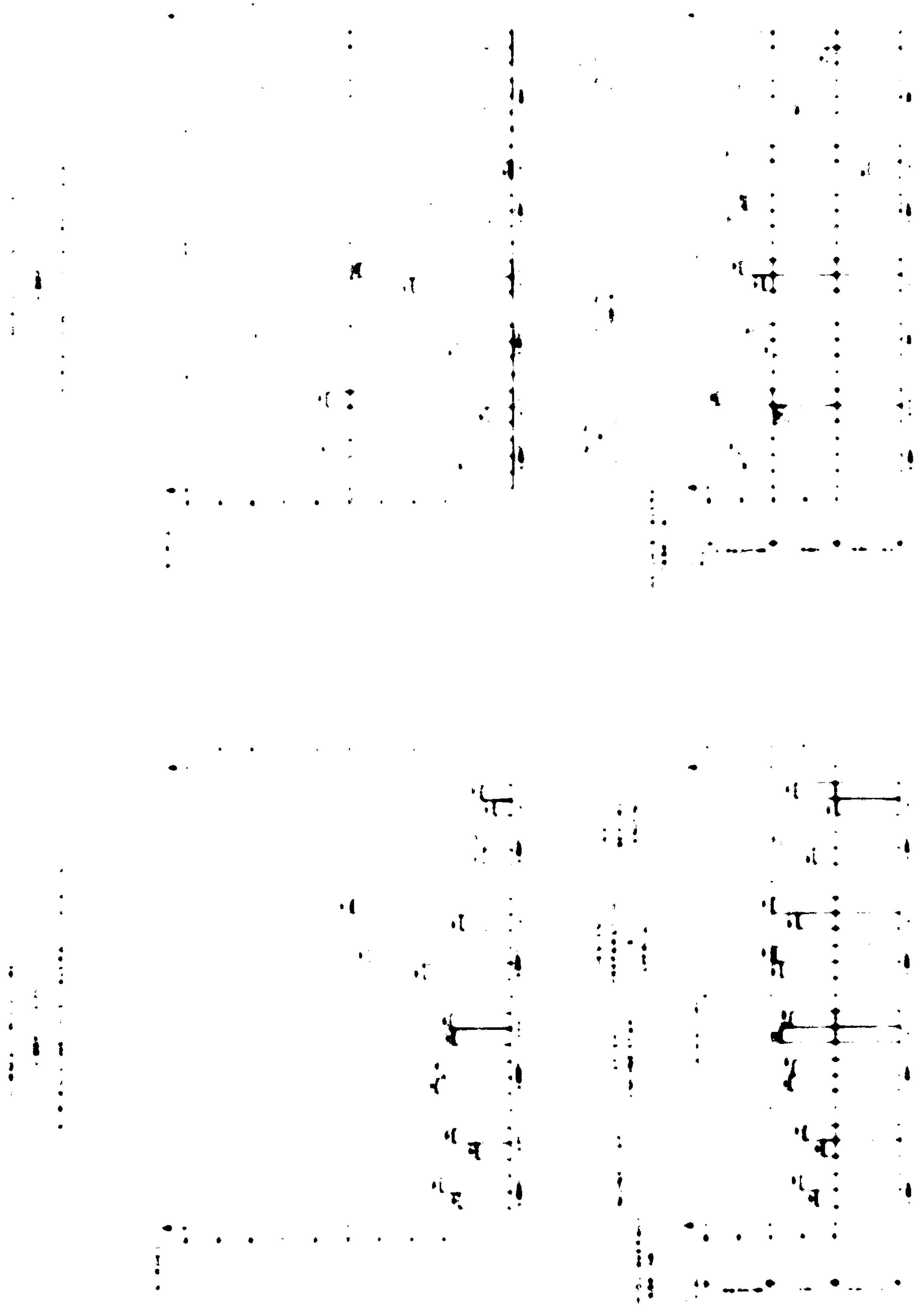


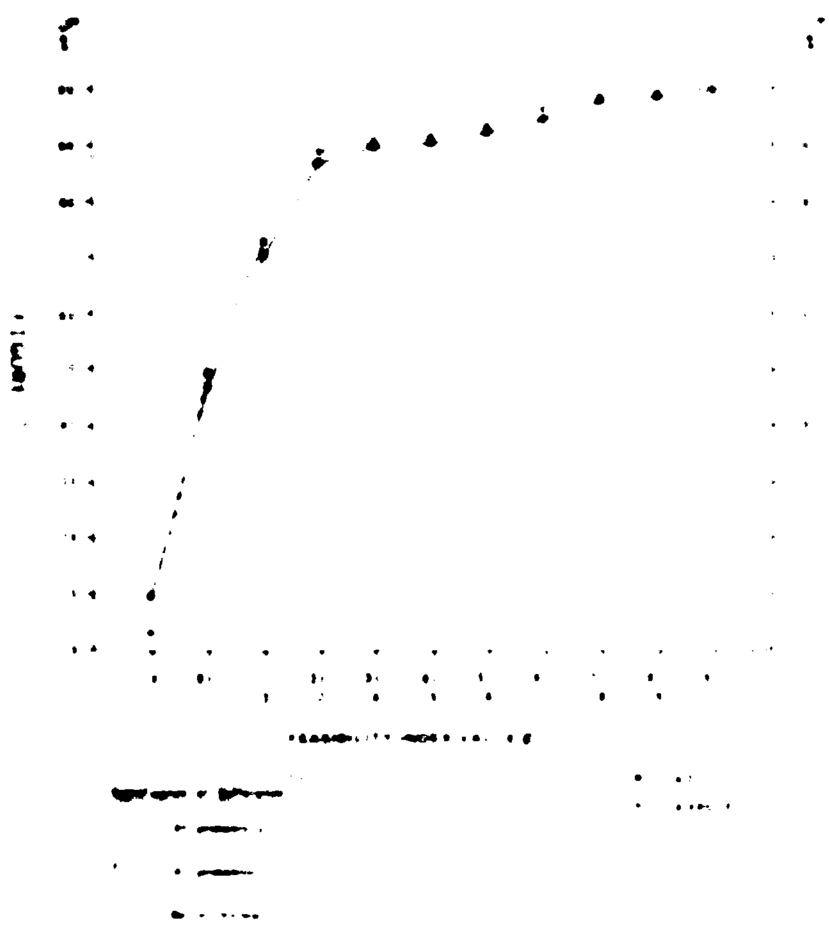




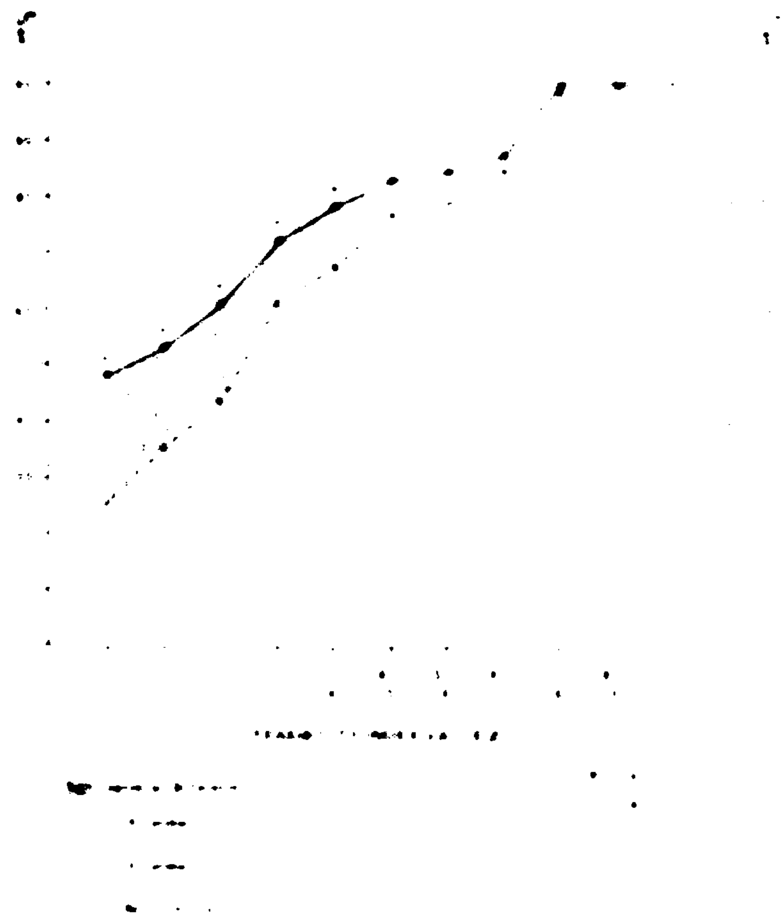




CM1 AND CM2 COMBINED
NORMAL PROCEDURES
VISIBILITY



CM1 AND CM2 COMBINED
ABNORMAL/EMERGENCY PROCEDURES
OBSERVABILITY



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DYNAMIC WORKLOAD ANALYSIS A300FF	
3 CREWS 4 OBSERVERS EUROPEAN NETWORK 15 SCENARIOS	
VALIDATION OF METHOD	
IN A300 SIMULATOR	44 h
WORKLOAD EVALUATION ON B 737	12 h
WORKLOAD EVALUATION ON DC 9	12 h
CREW TRAINING ON A300FF	
SIMULATOR	30 h
AIRCRAFT	6 h
WORKLOAD EVALUATION IN	
SIMULATOR CAMPAIGN ON A300FF	60 h
38 FLIGHTS	
WORKLOAD EVALUATION IN	
FLIGHT CAMPAIGN ON A300FF	60 h
96 FLIGHTS	
ATC WORKLOAD EVALUATION	
IN A300FF SIMULATOR	4 h

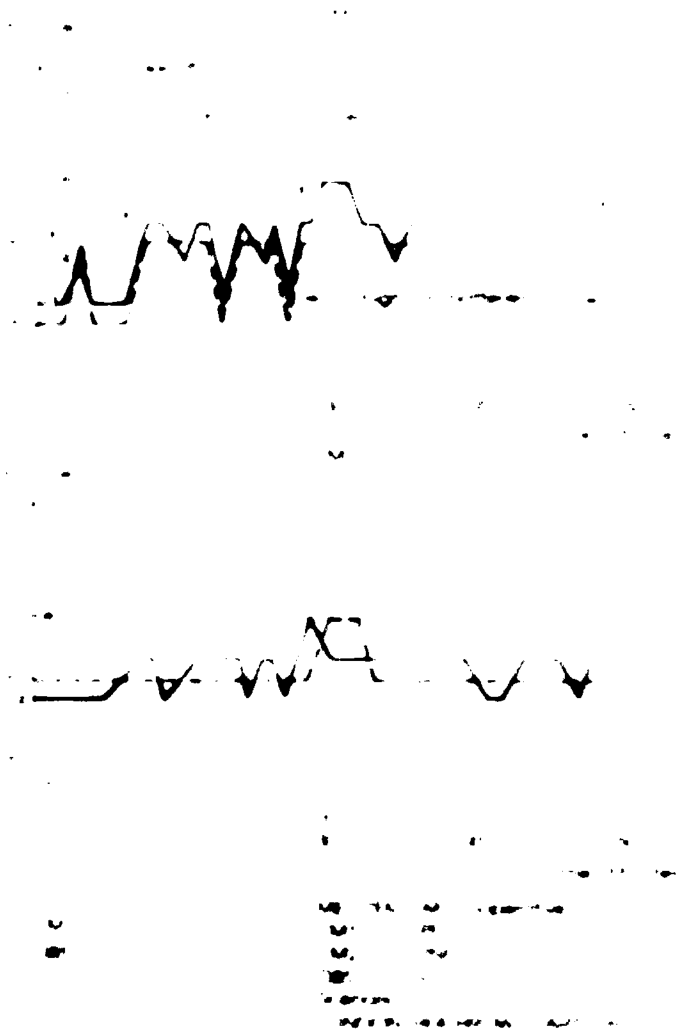
DYNAMIC WORKLOAD ANALYSIS A310

7 CREWS 8 OBSERVERS
EUROPEAN NETWORK LONG DISTANCE FLIGHTS
10 SCENARIOS

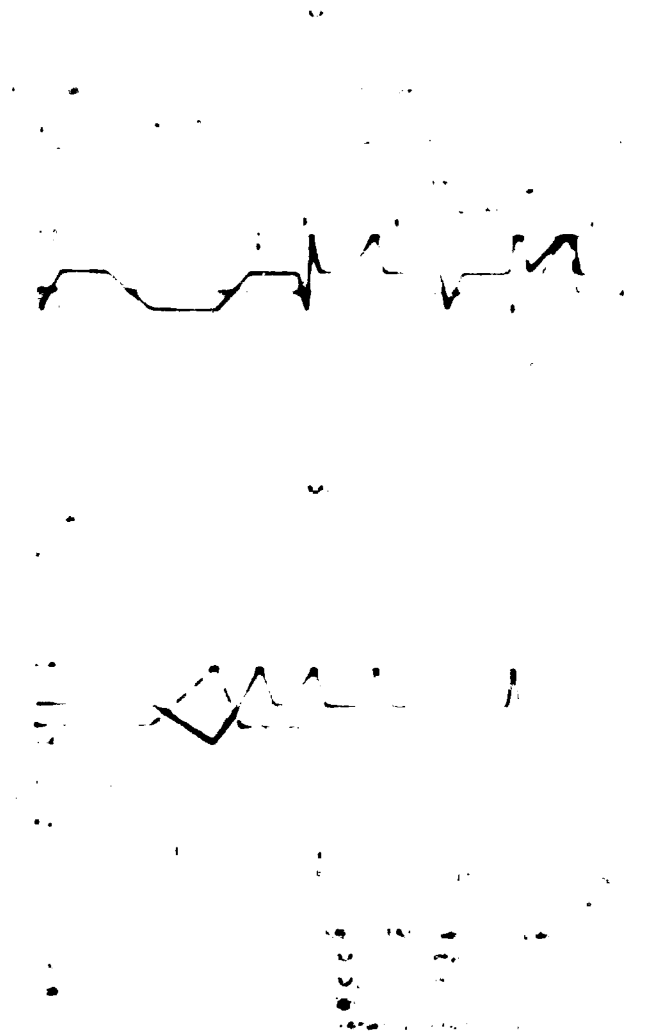
MINIMUM CREW CERTIFICATION		AIRLINE ORIENTED ANALYSIS	
3 CREWS WITH AA & PILOTS		4 CREWS WITH AL & PILOTS	
TRAINING TO METHOD		TRAINING TO METHOD	
IN A300FF SIMULATOR	4 h	IN B 737 OR BAC 111 SIMULATORS	6 h
CREW TRAINING ON A310		CREW TRAINING ON A310	
SIMULATOR	60 h	SIMULATOR	32 h
AIRCRAFT	6 h	AIRCRAFT	4 h
WORKLOAD EVALUATION IN		WORKLOAD EVALUATION IN	
SIMULATOR CAMPAIGN ON A310	48 h	SIMULATOR CAMPAIGN ON A310	16 h
162 FLIGHTS		72 FLIGHTS	
WORKLOAD EVALUATION IN		WORKLOAD EVALUATION IN	
FLIGHT CAMPAIGN ON A310	57 h	FLIGHT CAMPAIGN ON A310	27 h
43 FLIGHTS		24 FLIGHTS	

Fig. ... Dynamic Workload Analysis Programmes for A300FF and A310

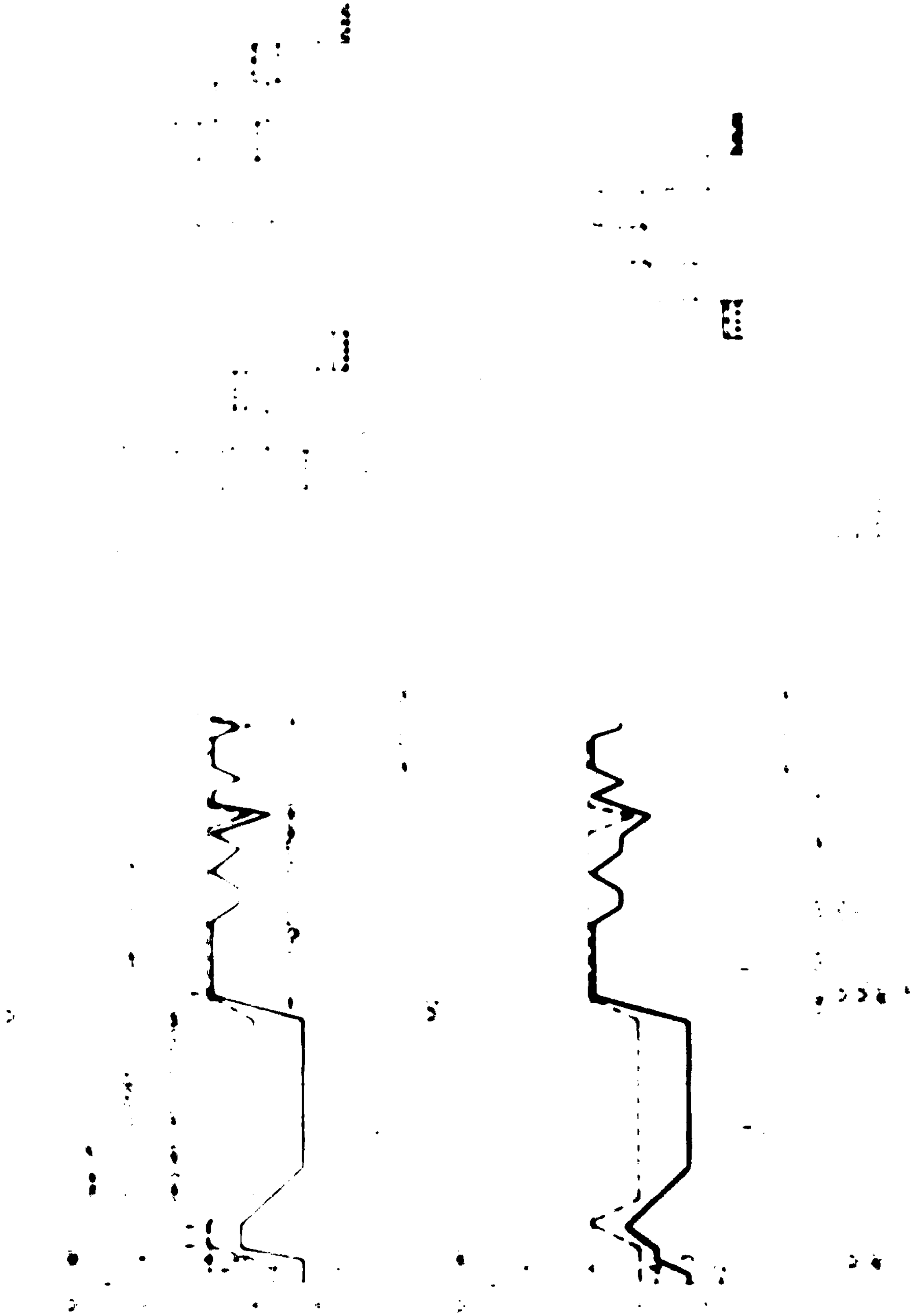
FIGURE [illegible]

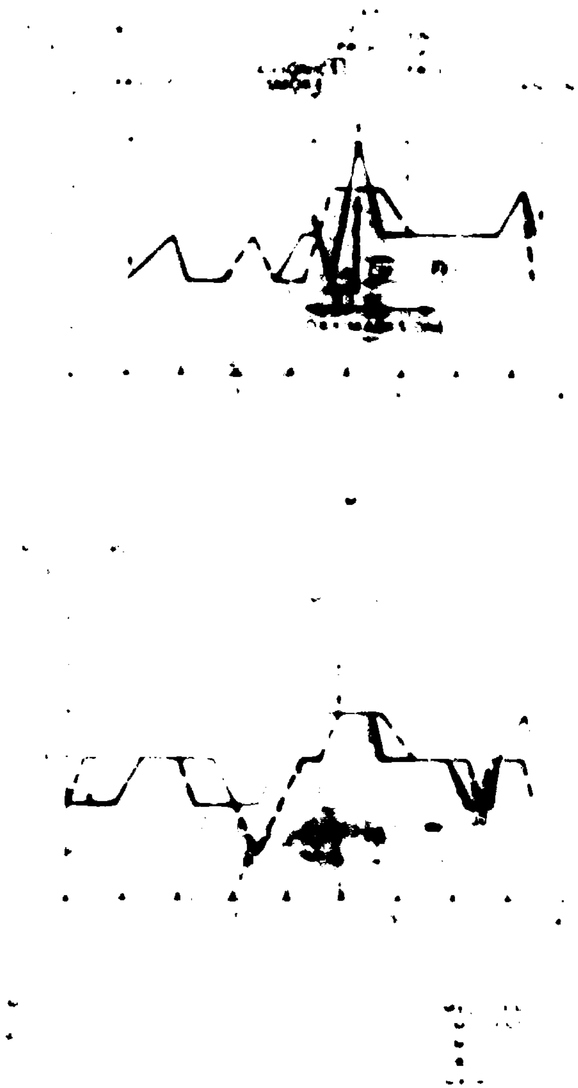


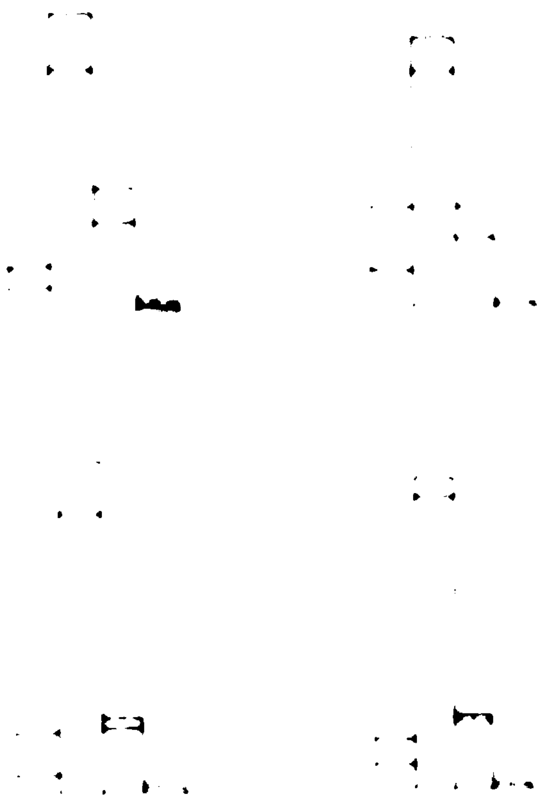
A [illegible] FT WORKLOAD TIMELINE
SIMULATOR CAMPAIGN

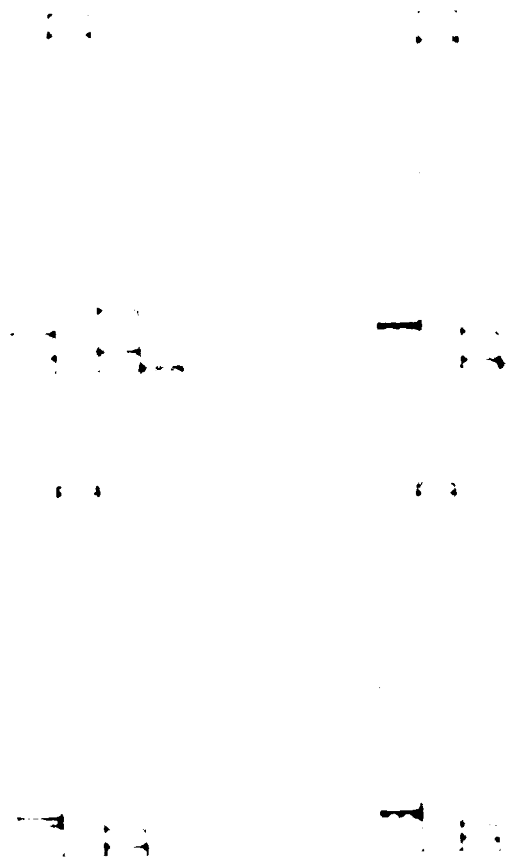


A [illegible] FT WORKLOAD TIMELINE
FLIGHT CAMPAIGN

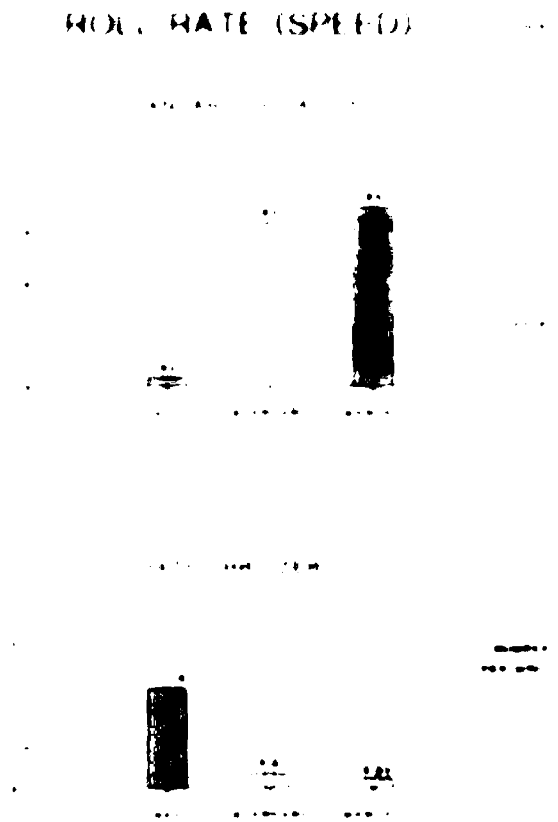


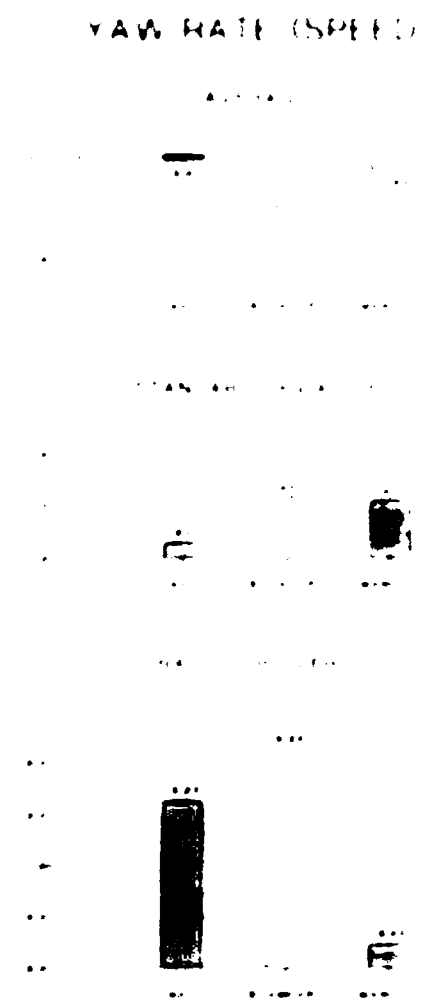


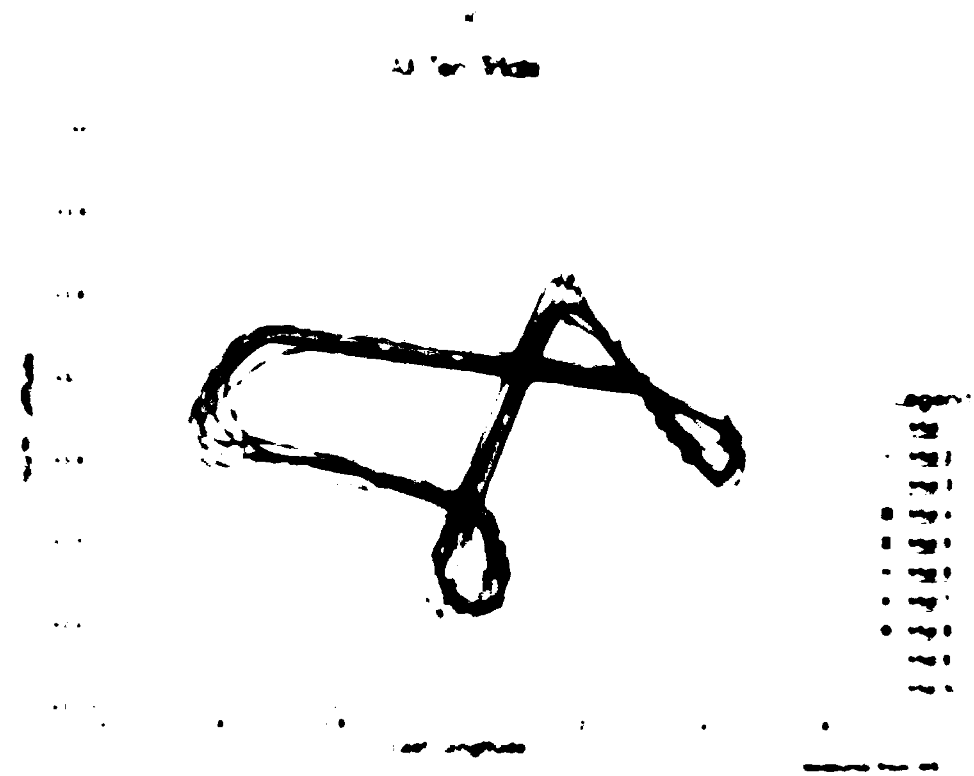


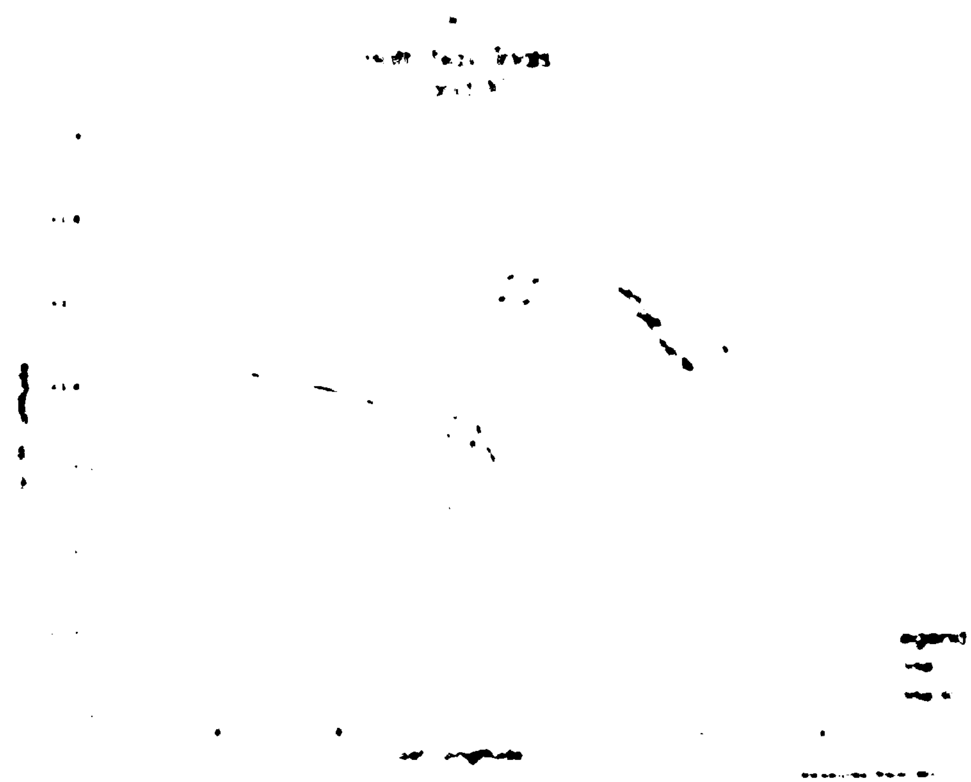


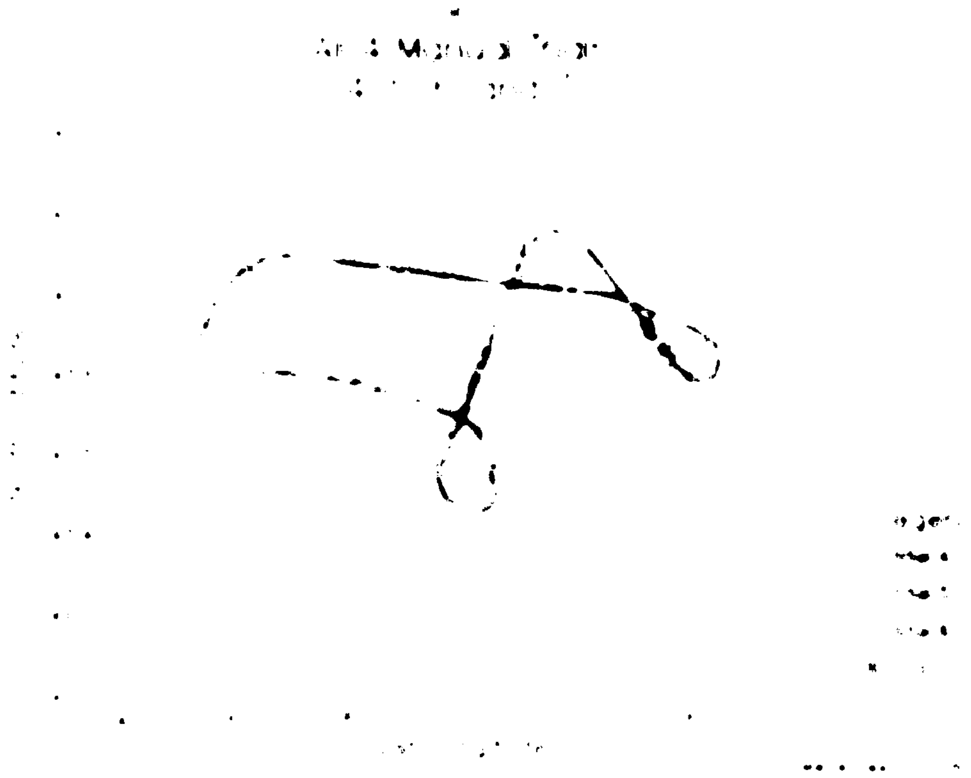

PITCH RATE (SPEED)


ROLL RATE (SPEED)


YAW RATE (SPEED)







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